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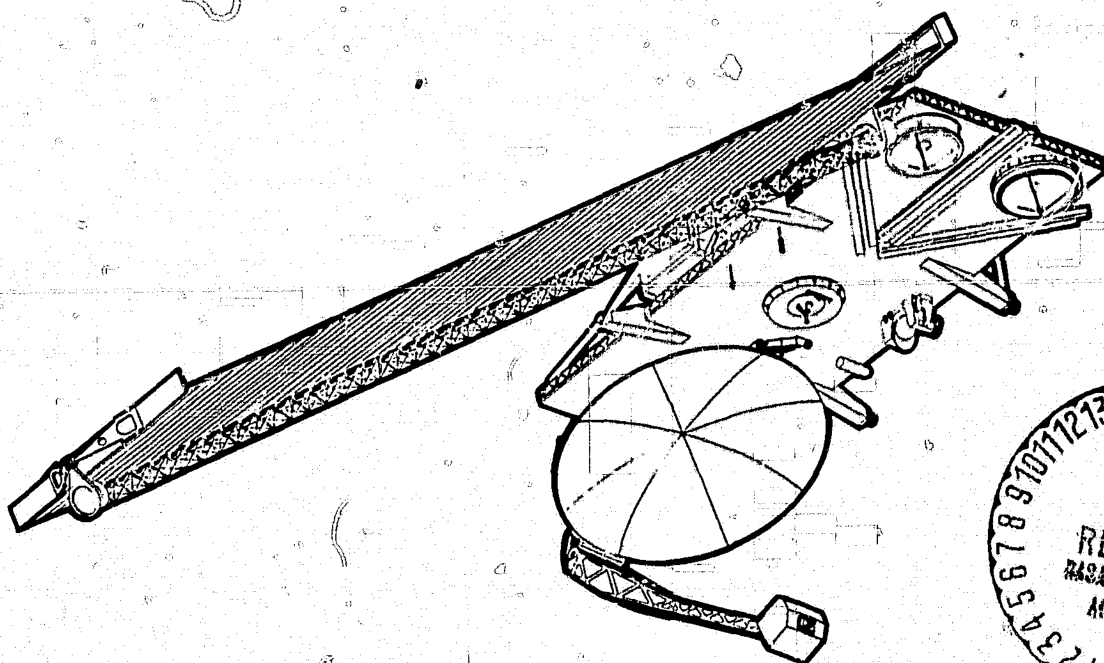
FLIGHT SEGMENT CONCEPT STUDY FINAL REPORT

NATIONAL OCEANIC SATELLITE SYSTEM



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LIST OF ACRONYMS

ACE	Attitude Control Electronics
ACS	Attitude Control Subsystem
ALT	Altimeter
C&DH	Communication and Data Handling
CDA	Command and Data Acquisition
CC (CM)	Center of Gravity (Center of Mass)
CITE	Cargo Integration Test Equipment
CSS	Coarse Sun Sensor
CU	Central Unit (C&DH)
CZCS	Coastal Zone Color Scanner
DOD	Department of Defense
DPU	Digital Processing Unit (GPS)
ES	Earth Sensor
EVA	Extra Vehicular Activity
FHST	Fixed Head Star Tracker
FSS	Fine Sun Sensor
GFE	Government Furnished Equipment
GPS	Global Positioning System
GSFS	Goddard Space Flight Center
HDR	High Date Rate
HCAS	High Gain Antenna
HPF	Hazardous Processing Facility
IF	Intermediate Frequency
IFB (I/F)	Interface Box
IR	Infrared
IRU	Inertial Reference Unit
JSC	Johnson Space Center
LAMMR	Large Amplitude Multichannel Microwave Radiometer
LRA	Laser Retroflector Array

MA	Multiple Access (TDRSS)
MACS	Modular Altitude Control System (MMS)
MMS	Multimission Modular Spacecraft
MPS	Modular Power System (MMS)
NDS	Navigational Data Satellite (GPS)
NOSS	National Oceanic Satellite System
NPS	NOSS Propulsion Subsystem
NSSC-1	NASA Standard Spacecraft Computer
OBC	On-Board Computer
PCU	Power Control Unit (C&DH)
PEP	Power Extension Package
PMP	Pre Modulation Processor
PPF	Payload Processing Facility
PPR	Payload Processing Room
PSDCU	Power Switching, Distribution and Central Unit
RIU	Remote Interface Unit
RMS	Remote Manipulator System
RPA	Receiver Processor Assembly
SCAT	Scatterometer
SLO	Solid State Local Oscillator (SCAT)
SMM	Solar Maximum Mission
SSA	S-Band Single Access
STACC	Standard Telemetry and Command Component
STDN	Standard Tracking & Data Network
STINT	Standard Interface Unit
STS	Space Transportation System
TDRS	Tracking and Data Relay Satellite
TDRSS	Tracking and Data Relay Satellite System
TM	Telemetry
TR	Tape Recorder
TWT	Traveling Wave Tube
TWTA	Traveling Wave Tube Amplifier
VAFB	Vandenburg Air Force Base
WTR	Western Test Range

1.0 INTRODUCTION

1.1 BACKGROUND AND PURPOSE

The GSFC National Oceanic Satellite System (NOSS) project was established in the spring of 1979 and the efforts necessary to start a project were initiated (Preliminary Execution Phase Project Plan, schedules, budgets, etc.). At that time, everyone on the project was new to the NOSS concept and the main source of concept definition was the Tri-Agency NOSS system study (23 March 1979). Programmatic changes were occurring, subsequent to this report, and the project needed a baseline document for technical definition of the flight segment.

In addition to these changes, the GSFC project personnel were unfamiliar with the instrument and the interactions between the instruments and the other spacecraft subsystems.

It was decided that an in-house conceptual design study would be the most effective means to learn the intricacies of the NOSS instruments and spacecraft problems as well as the interactions with other systems (STS, TDRSS, etc.).

Thus, in June, project approval was given to conduct a low-level, in-house conceptual design for a NOSS spacecraft. This design was not intended to be the preferred system design nor an optimized design, but rather a design which would show feasibility and familiarize the project personnel with the NOSS mission by requiring a comprehensive look at all spacecraft areas. The study began in July 1979 and was completed at the end of October 1979. Use of the multimission modular spacecraft (MMS) modules was adopted as a matter of expediency due to the limited resources available to perform the study. There is neither a requirement to use MMS equipment for NOSS nor is there a project preference for the MMS equipment.

This report is a result of that effort. The remainder of this section will give the guidelines used for the study. Section 2 is a basic description of the overall spacecraft concept. The following sections describe the instruments (Section 3), the spacecraft subsystem (Section 4), and the mission timelines (Section 5). Section 6 is a summary of the major items that have been brought to light by the study. Finally, the detailed drawings and block diagrams appear as Section 7.

1.2 STUDY REQUIREMENTS AND GUIDELINES

Table 1.2-1 is a list of the main guidelines given at the start of the study. In addition, a type of design was suggested wherein the spacecraft should be basically a large flat structure that would stow directly in the shuttle. It was felt that this would maximize the earth viewing area and minimize the number of deployments.

The orbit was arbitrarily chosen but is in the middle of the range of those orbits under consideration at the time.

TABLE 1.2-1

STUDY GUIDELINES

- Sun Synchronous Orbit 800 Km, 10:30 AM Asending Node
- Spacecraft Location Knowledge to ± 400M in X, Y, Z
- Pointing Accuracy to $\pm$ 0.2^{o}, knowledge onboard to $\pm$ 0.1^{o}, on the ground to $\pm$ 0.03^{o} (3σ)
- STS Launch from WTR to 300 Km, mission orbit inclination
- On-orbit propulsion for transfer from 300 to 800 Km and for time adjustment (in-plane) only. No plane change capability.
- Five-Year System Operation
- Three-Year Design Life Goal
- Instrument Complement of:

  2 Altimeters (ALT) (For Redundancy)

  1 Scatterometer (SCAT)

  1 Large Antenna Multichannel Microwave Radiometer (LAMMR)

  1 Coastal Zone Color Scanner (CZCS)
- All instruments on continuously except CZCS, which operates on a 25% Duty Cycle, except for mirror drive and dynamics compensation which are operated on a 100% Duty Cycle.
- 25% Contingency for instrument growth
- TDRSS used for all T&C and Data Communications
- GPS used for spacecraft on-board position determination
- Design spacecraft to be retrievable but not reserviceable in orbit.
- TDRSS S-Band Single Access availability is 25%.

2.0 SPACECRAFT DESCRIPTION AND INTERFACES

2.1 SPACECRAFT DESCRIPTION

Figures 2.1-1 and 2.1-2 are two views of the design that evolved during the study. The spacecraft weighs about 14,469 pounds, is 11 feet wide, and 26.5 feet long. The locations of the major components are shown on the figures.

It will be noted that there are a large number of drawings in this report. A good deal of thought went into the sizing and location of the equipment for this spacecraft in an effort to assure that no major items were overlooked. All the drawings are to scale and the equipment locations are chosen to facilitate the instrument operations. Many other factors have also been taken into consideration such as the location of the hydrazine tank at the spacecraft CG and the thruster locations. Most of these considerations will be discussed in the sections pertaining to the various subsystems.

The thermal control of the spacecraft will entail blanketing most of the boxes and structures but these blankets have been left out of the drawings for clarity.

2.2 STS INTERFACE

The basis of this spacecraft is a long flat structure that allows maximum instrument viewing area and yet can still be stowed directly in the shuttle. Figure 2.2-1 shows the spacecraft stowed in the cargo bay. The four trunnions mount directly to the standard sill fittings. A single keel fitting is provided on the structure below the propulsion tank.

Current planning is for the minimum number of electrical connections between the shuttle and the spacecraft. It would be checked out via an umbilical at the launch site and the umbilical then would be removed. The only connections to the shuttle would be for a trickle charge on the batteries and the caution and warning telemetry required to confirm status of safing for the STS crew.

NOSS

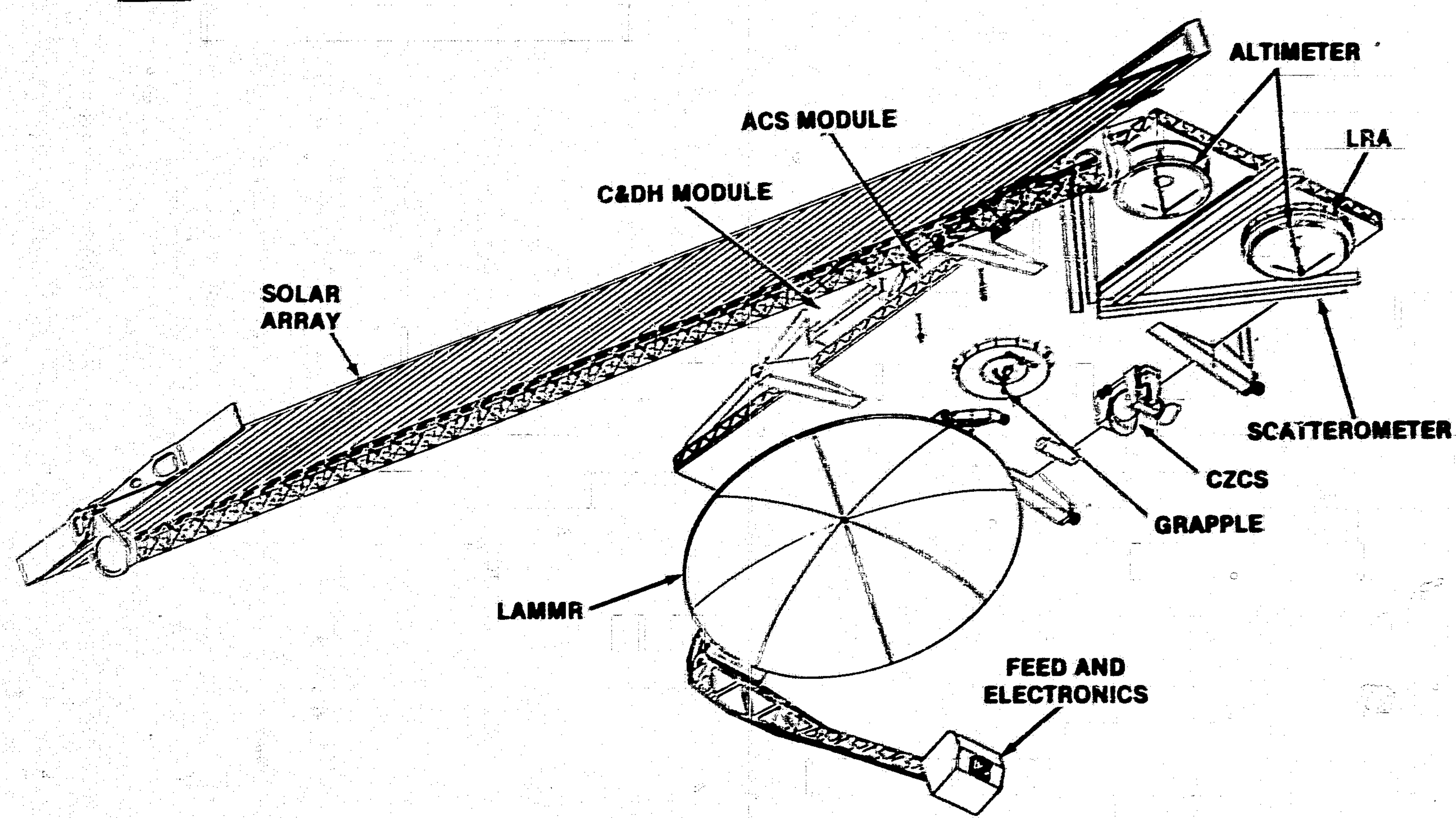


Figure 2.1-1. Earth Viewing Side Of NOSS Spacecraft - On Orbit Configuration

NOSS

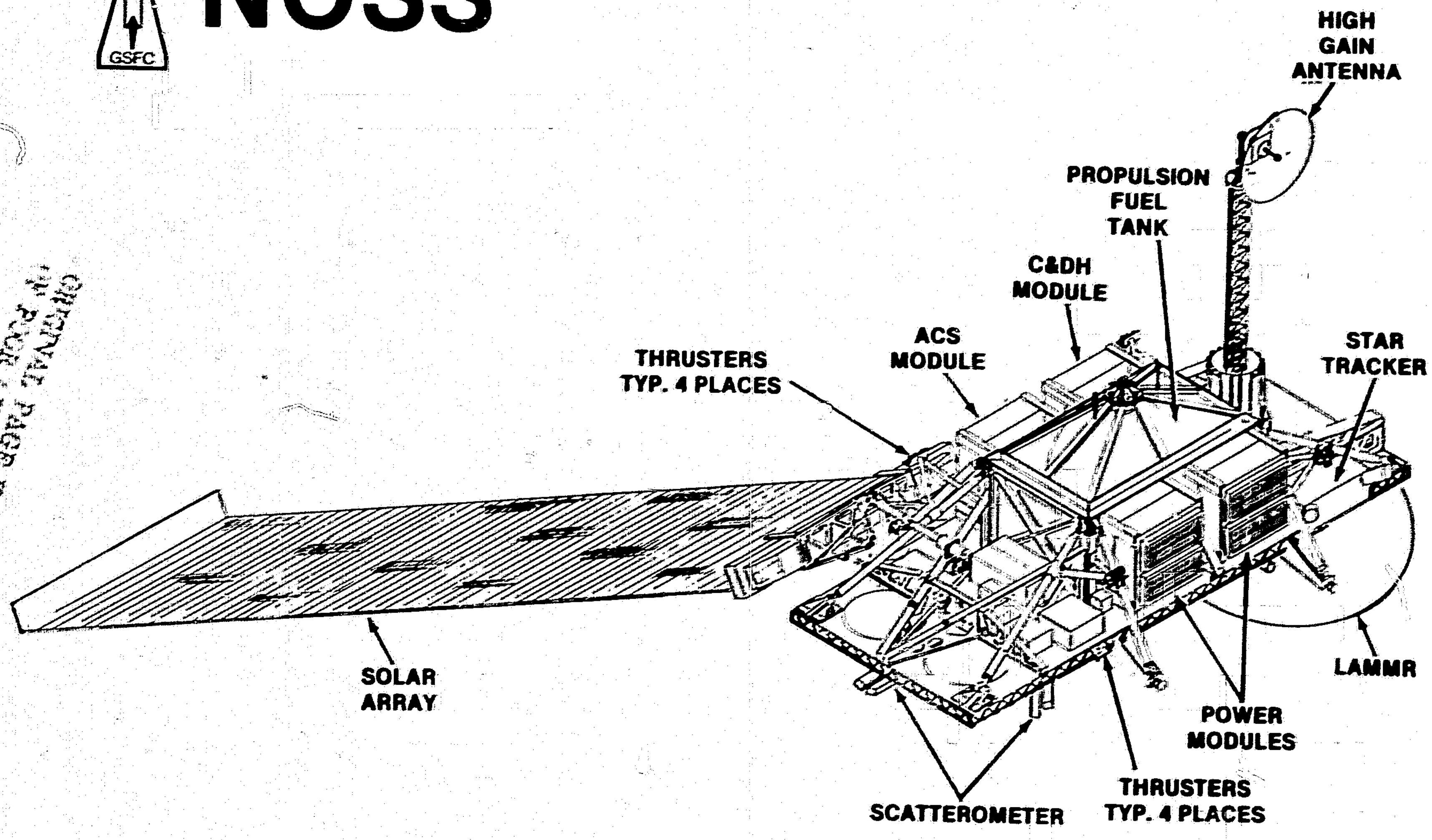


Figure 2.1-2. Space Viewing Side Of NOSS Spacecraft - On Orbit Configuration

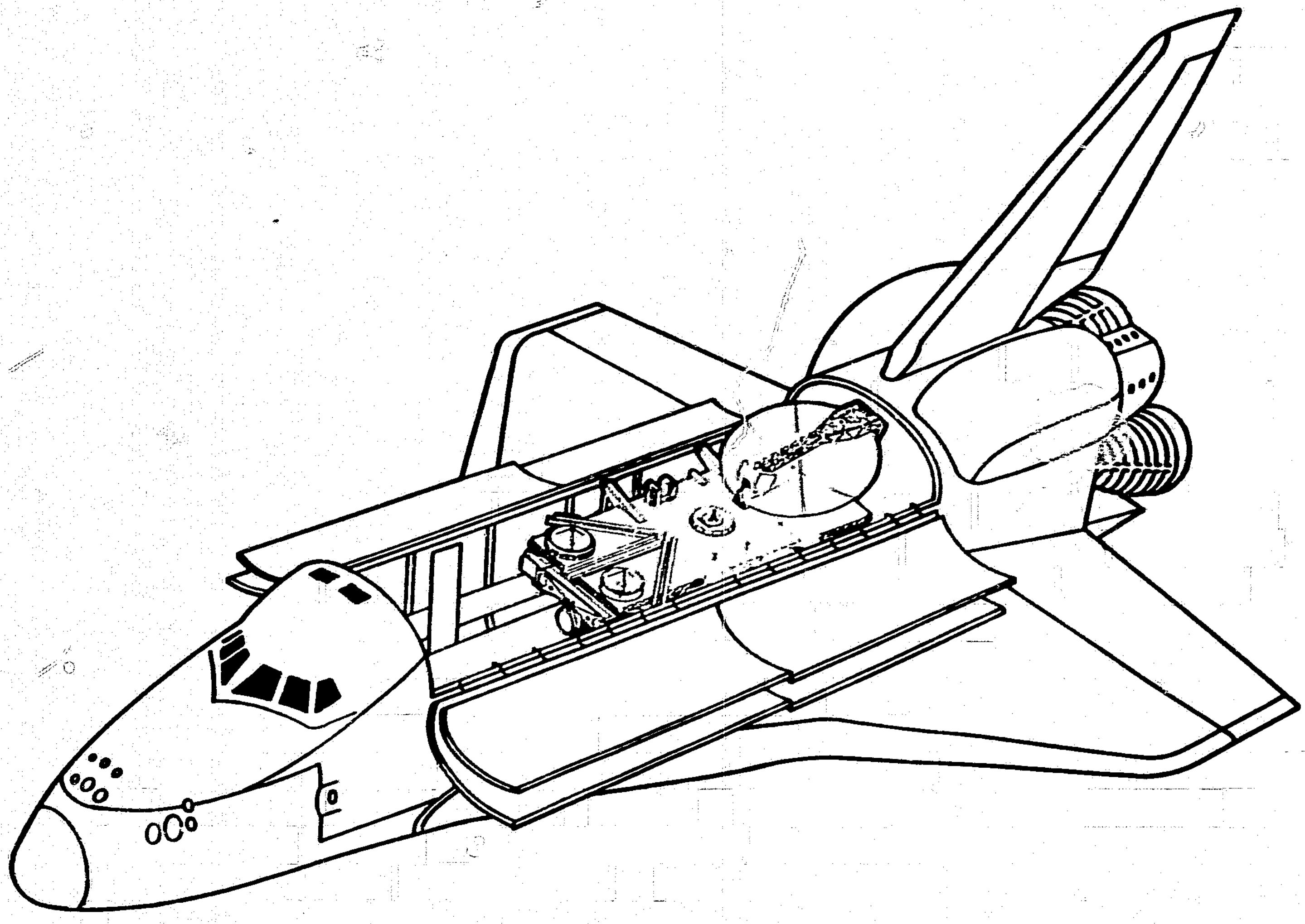


Figure 2.2-1. NOSS Spacecraft Stowed In Orbiter Payload Bay

Upon reaching the shuttle orbit (~ 300 KM), the RMS would be used to remove the spacecraft and orient it to the appropriate attitude. Spacecraft checkout then commences via TDRSS.

After initial checkout, the deployments would be initiated (solar array, high gain antenna, LAMMR feed). This deployed configuration is shown in Figure 2.2-2. Since retrievability is required, all deployments must be reversible, hence motors must be used and rapid motion can be avoided. The spacecraft is then released by the RMS and moves slowly away from the shuttle. When a safe distance is achieved, orbital transfer operations commence to move the spacecraft to the mission orbit of 800 KM.

2.3 TDRSS INTERFACE

All the telemetry, command, and data transmissions are presumed to be via TDRSS at S-band. The Multiple Access (MA) return services will be used continuously. The S-band Single Access (SSA) service will be used for high data rate tape recorder dumping about 15 to 20 minutes per orbit. The SSA forward service for command will also be used during this same 15 to 20 minute period.

Telemetry and command can be accomplished at a low rate via the S-Band omni antenna system on NOSS and the SSA system on TDRSS. Telemetry, command, and lower rate data will normally be accomplished via the NOSS high gain antenna and the TDRSS MA system. High data rate type dumping will be via the NOSS high gain antenna and the TDRSS SSA system. The equipment has been configured so that all telemetry, command and data functions can be accomplished directly by a ground station as backup.

This backup has not been analyzed in detail, since the ground sites and the frequencies may change depending on programmatic decisions. In addition, the use of the K-band service on TDRSS to replace S-band for high data rate tape dumping has been considered and does not appear to have significant effects on weight or power.

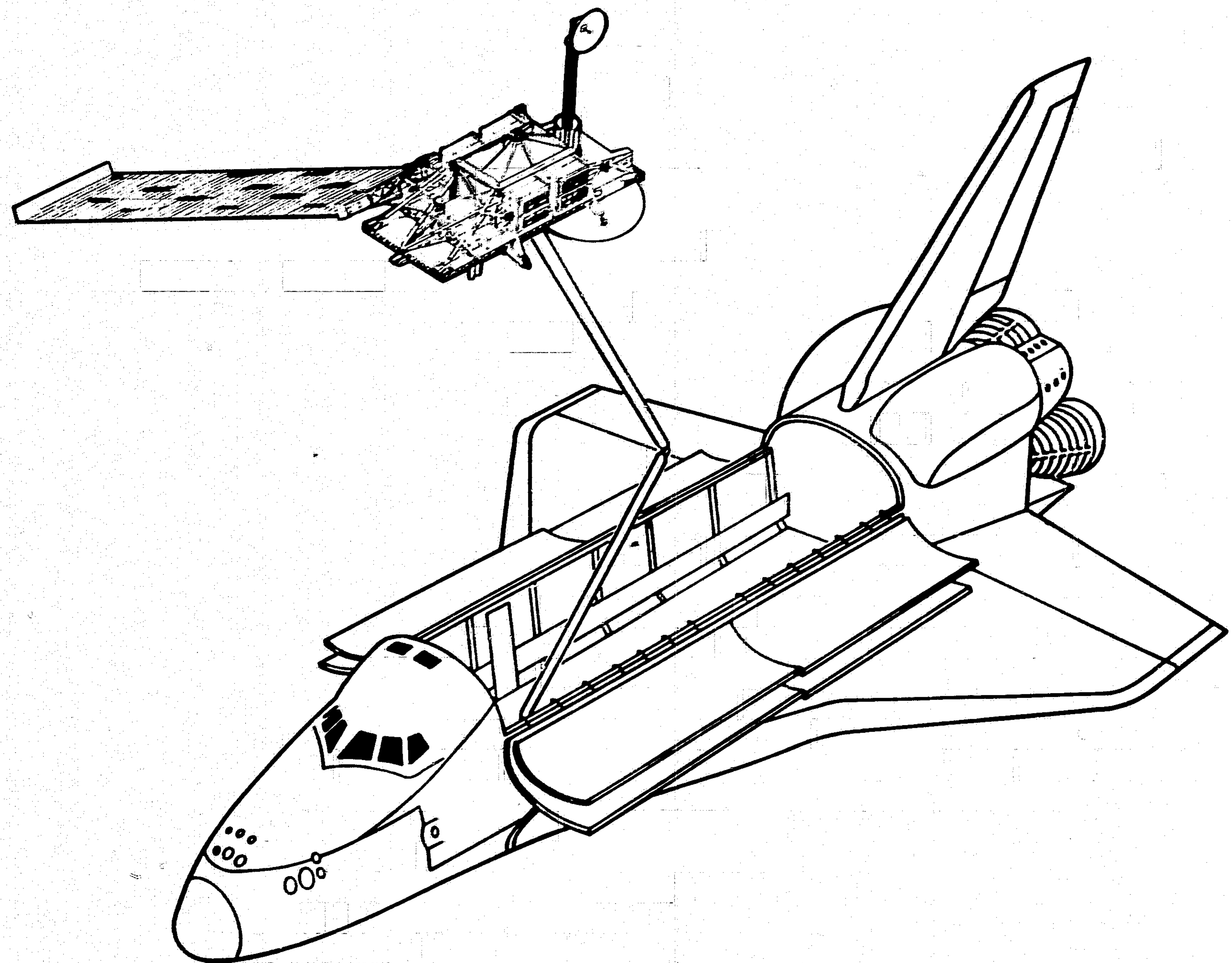


Figure 2.2-2 NOSS Spacecraft Deployed On RMS

2.4 SUBSYSTEMS AND INSTRUMENTS

The basic instrument complement for NOSS consists of three previously flown instruments and one now in development. The new instrument is the Large Antenna Multichannel Microwave Radiometer (LAMMR) being developed at GSFC. It is a much larger version (3.6 meter diameter) of the Scanning Multichannel Microwave Radiometer (SMMR) flown on NIMBUS-7. Primary usage is the monitoring of the sea surface temperature, wind speed, sea ice and atmospheric corrections for the Altimeter and Scatterometer.

The Altimeter (ALT) is the same instrument developed by NASA Wallops Island for SEASAT, except that two are being flown on NOSS for redundancy to meet life requirements. Primary usage is the monitoring of ocean wave parameters and possibly ocean currents.

The Scatterometer (SCAT) is an upgraded version of the SEASAT-A Scatterometer System (SASS). The NOSS version will have six antennas instead of four and also redundant electronics. This instrument has been developed by NASA Langley Research Center. Primary usage is the monitoring of wind velocity over the oceans.

The Coastal Zone Color Scanner (CZCS) is the same instrument that was developed by GSFC and flown on Nimbus-7 except that three additional channels have been added. Primary usage is the monitoring of chlorophyll concentration and water turbidity distributions.

A GPS system similar to the one on LANDSAT-D will also be used on NOSS for on-board determination of spacecraft location and velocity. While this is not a scientific instrument, it is treated the same as the other instruments for interface purposes.

Figure 2.4-1 is a basic block design of the spacecraft systems. Since most of the instruments have flown on different spacecraft, the interfaces are different and an interface box has been attached to each instrument to condition the power, telemetry and command and data for that particular interface. Section 3 will describe these instruments and their operation in detail.

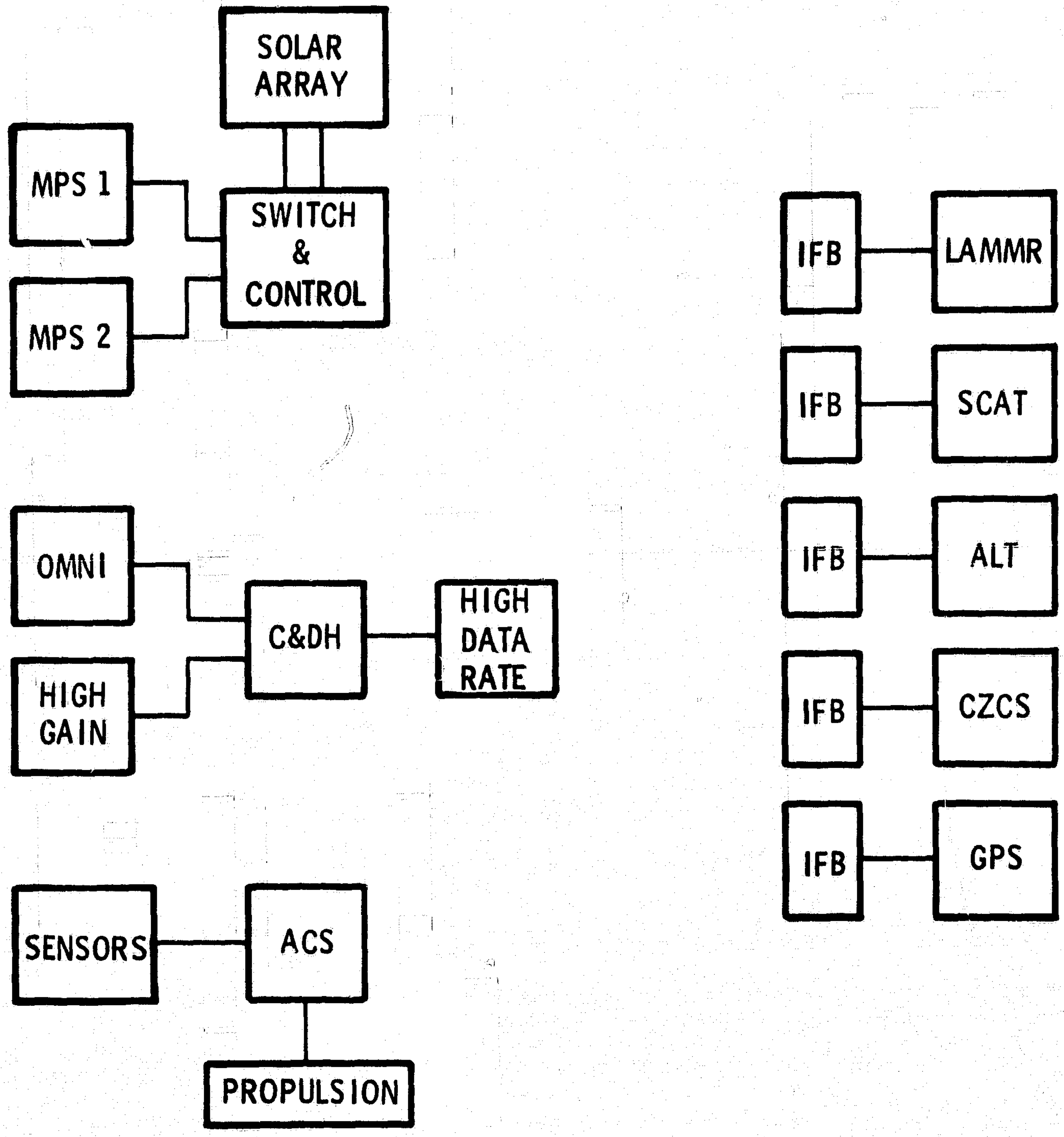


Figure 2.4-1. Basic Block Diagram

The MMS modules were chosen as a basis for this design although the basic MMS bus configuration was not used. The power subsystem requires two modular power system (MPS) modules. In addition, there is enough extra equipment needed to control both MPS modules, the solar array, and power switching functions that an additional box (power switching and control) is required to house these components.

The C&DH system is based on the MMS C&DH module with some additions. These consists of a high gain antenna unit, high data rate S-band transmitters, a high data rate handling system, four large tape recorders, and encryption/decryption equipment. As with the power system, an additional box was necessary to contain most of this equipment and is called the high data rate box.

The ACS system uses the MMS module but the star tracker and other sensors are mounted externally.

2.5 SPACECRAFT MASS, POWER AND COMMUNICATION BUDGETS

In Table 2.5-1, the budgets for the spacecraft power, mass, telemetry, command and data are summarized. A 25% instrument growth was assessed for all budgets and a 15% contingency was added for spacecraft mass. Without contingency, the current mass estimate is 5706 kg (12,580 lbs) and with contingency, this becomes 6,562 kg. (14,469 lbs). These values brackets the weight estimates used for the preliminary mass properties (Section 4.5) analysis and propellant sizing (Section 4.2).

Power budgets include the 25% Instrument growth. Power levels quoted are orbital average.

Command and telemetry requirements for the instruments and unique equipment have not been addressed in this study but a large margin exists between the C&DH module capabilities and the totals listed in the table.

The NOSS mission lift-off weight margin is analyzed in Table 2.5-2. Significant margin (4,338 lbs) is shown for the anticipated shuttle lift-off capability.

TABLE 2.5-1

SPACECRAFT MASS, POWER AND COMMUNICATION BUDGETS

SUBSYSTEM	MASS KG	POWER WATTS	TELEMETRY ANALOG	TELEMETRY DIGITAL	DATA KBS	COMMAND PULSE	COMMAND LEVEL	COMMAND SERIAL
STRUCTURAL/THERMAL	1460	400	44	-	-	18	-	-
POWER	712	170	142	68	-	196	-	-
PROPULSION								
Dry Weight	269	135	41	49	-	52	20	-
Propellant	1724							
ATTITUDE CONTROL	198	156	69	26	-	57	-	6
TELEMETRY, COMMAND & DATA	340	243	74	170	-	114	-	15
INSTRUMENTS								
LAMMR	320	350	35	100	64	30	-	-
SCAT	180	165	-	-	4	18	12	-
ALT	200	177	14	-	8.5	16	8	-
CZCS	50	50	50	30	1200 (AVG)		40	-
LRA	15	-	-	-	-	-	-	-
I/F BOXES	38	116	-	100	-	200	-	-
25% INSTRUMENT GROWTH	200	215	25	58	32	66	15	-
SUBTOTAL	5706	2177	494	601	1622.5	767	95	21
Contingency	856	32						
LAUNCH TOTAL	6562	-						
ON-ORBIT TOTAL	5763	2209	494	601	1622.5	767	95	21
RETRIEVAL TOTAL	4958	-						

Table 2.5-2

Mission Lift-Off Weight Margin

	Wgt. (kg)
NOSS Spacecraft	6562
NOSS Airborne Support Equipment	100
STS Payload Chargeable Items	364
TOTAL	7026
STS Payload Capability	11,364
Lift-Off Margin	4,338

3.0 INSTRUMENT DESCRIPTIONS

3.1 MISSION SCIENCE

The National Oceanic Satellite System (NOSS) will yield a fast, world-wide look at what is occurring on or just above the surface of the sea by pointing four instruments at the world's oceans. Three are microwave instruments and one is a visible/IR radiometer. Tables 3.1-1 and 3.1-2 summarize the instrument characteristics and Figure 3.1-1 shows instrument locations.

The three NOSS microwave instruments provide information that can be translated into sea-surface temperature, wind speed and direction, wave heights, sea ice features, ocean topography, and atmospheric water. Ocean waves, ice fields, icebergs, ice leads (linear openings in the ice through which ships may navigate), and sea conditions along the coastlines will be charted. Significant waveheight (the average of the largest third of the ocean waves) will be measured. Spacecraft altitude above the sea surface will be measured within 10 cm, allowing determination of ocean tides, storm surges, and currents.

The NOSS visible/IR radiometer will chart cloud cover and ocean and coastal features, and measure the ocean color. One of the NOSS instruments is a new development and three are minor modifications of instruments previously flown on other spacecraft. The current configurations of all instruments were assumed for the NOSS study. Additional studies are under way to determine the actual flight configurations.

The radar Altimeter (ALT) height measurement is determined from the round-trip time required for a radar pulse to travel to the surface of the ocean and return to the altimeter. Height measurement data are related to ocean circulation, storm surges, tides, and wind-driven, sea-surface tilts. The return pulse shape provides a measure of wave height parameters. Ocean radar backscatter coefficient is an indication of the reflectance properties of the ocean surface and can be related to the magnitude of near-surface winds.

The Large Antenna Multichannel Microwave Radiometer (LAMMR) is a passive microwave sensor in which radiometers measure antenna temperatures that

Table 3.1-1
BASELINE NOSS INSTRUMENTS

SENSOR	ALTIMETER	SCATTEROMETER	LARGE ANTENNA MULTI CHANNEL MICROWAVE RADIOMETER	COASTAL ZONE COLOR SCANNER	GLOBAL POSITIONING SYSTEM
ACRONYM	ALT	SCAT	LAMMR	CZCS	GPS
DEVELOPMENT	IDENTICAL TO SEASAT	MOD OF SEASAT SASS	PHASE B DESIGN STUDY	MOD OF NIMBUS -7 CZCS	SIMILAR TO LANDSAT-D
FREQ/λ	13.5 GHz 2.2 cm	14.6 GHz 2.0 cm	4.3, 10.65, 18.7, 21.3, 36.5 GHz	440, 520, 560, 640, 685, 750, 880 nm; 10.8, 12.0 μm	1575.42 MHz 1227.60 MHz
MEASUREMENT	TRAVEL TIME, PULSE FORM DISTORTION, SURFACE REFLECTIVITY	RADAR BACKSCATTER	MICROWAVE BRIGHTNESS TEMP	VISIBLE & IR RADIANCE	PROPAGATION DELAY, DOPPLER
FOOTPRINT, km	23	50	7 to 40	0.7	-
PARAMETER / RESOLUTION	ALT, 10 cm; H-1/3 1m; OCEAN TIDES CURRENTS	WIND VEL, 2m/s WIND DIR, 20^o SEA ICE	WATER TEMP, 1^oK WIND, 2m/s WATER VAPOR SEA ICE ICE AND SNOW	CHLOROPHYLL PLANKTON SEDIMENT SURFACE TEMP	POSITION VELOCITY TIME
SCAN	NADIR	FAN BEAMS 45^o, 90^o 135^o BOTH SIDES OF TRACK	43^o CONE, 360^o SCAN	$\pm 40^o$ CROSS TRACK	OMNI

Table 3.1-2

INSTRUMENT PHYSICAL CHACTERISTICS

SENSOR	ALT	SCAT	LAMMR	CZCS	GPS	TOTAL
WT, Kgm	200	180	320	50	17	767
POWER, W	177 (ALT #2 OFF)	165 400 (peak)	350	50 85 (peak)	35	777
ENVELOPE, cm	2(34x51x25) 2(100 x 80)	115 x 55 x 31 6(300x10x15)	140 m^3	85 x 40 x 60	31 x 42 x 21	
DATA RATE, kbps	8.5	4	64	5250	1	5262

Includes GPS but not interface boxes or LRA.

NOSS

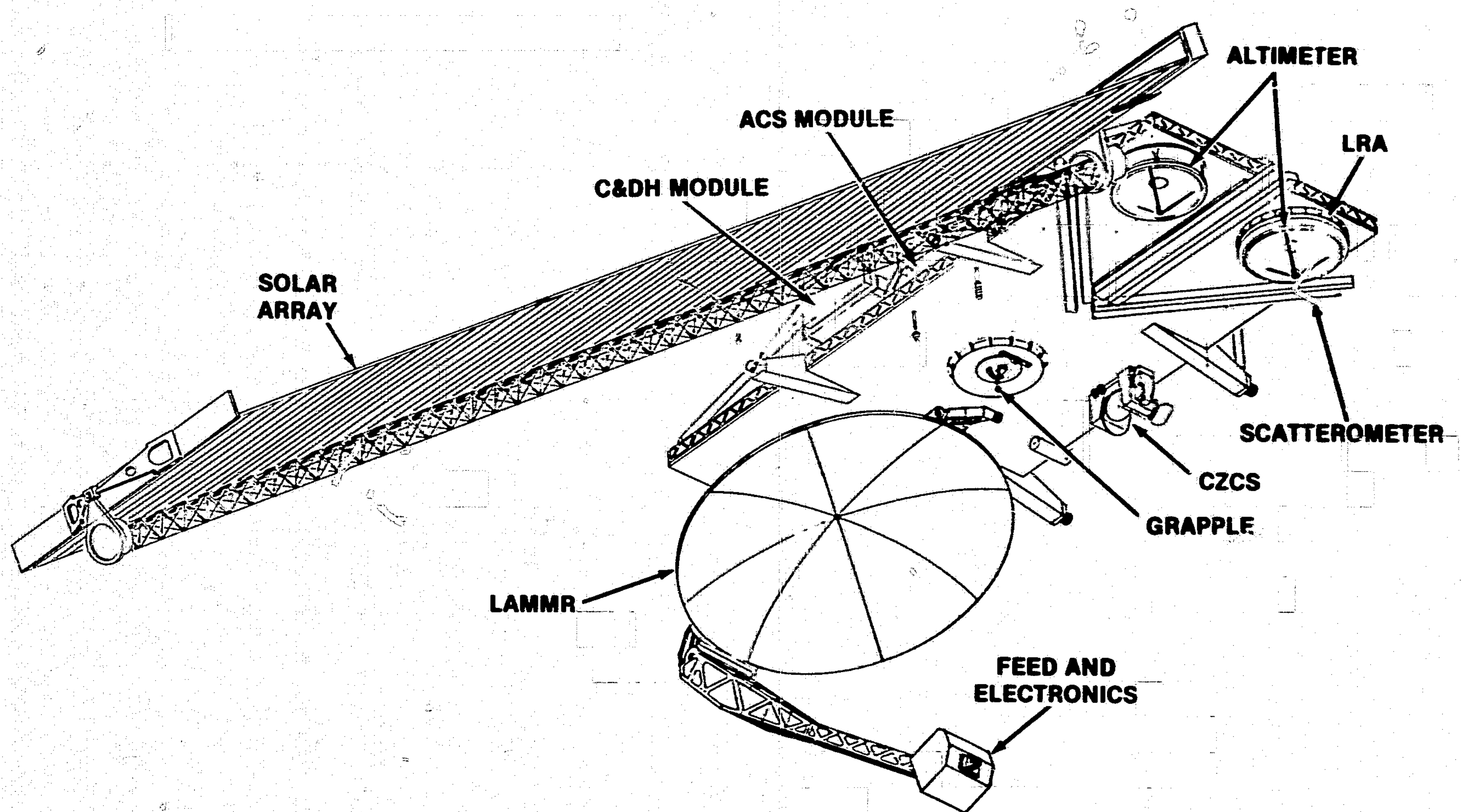


Figure 3.1-1 Instrument Layout

are the weighted integral of the ocean brightness temperature distribution over all solid angles. The antenna pattern effects are inverted, and brightness temperatures are derived from these measurements. Models of ocean surface emissivity and atmospheric emission and absorption are used to derive estimates of ocean surface temperature, wind speed, sea ice concentration and type, and atmospheric water content. Brightness temperature and atmospheric estimates are used to correct the ALT and SCAT measurements.

The Scatterometer (SCAT) measures radar backscatter coefficients which are converted to wind direction and magnitude by combining three measurements of radar backscatter for each resolution cell and applying one of several models to the measurements. Backscatter dependence on wind speed derives from the effect of wind on the ocean surface capillary wave spectrum, and is affected by small surface disturbances such as those induced by falling rain. Microwave brightness measurements from LAMMR are used to correct SCAT backscatter measurements for the effects of atmospheric attenuation.

The Coastal Zone Color Scanner (CZCS) measures the energy received from the sea in the visible, near infrared, and infrared spectral ranges. Seven visible channels of the scanning radiometer measure the solar energy and, therefore, water color, as affected by absorption and scattering due to chlorophyll, sediment, and gelbstoffe (yellow substance). Two other channels in the infrared measure the temperature of coastal waters and the open ocean. All instruments operate continuously except CZCS, which has an approximate 25 percent duty cycle.

The spacecraft can accommodate an additional complement of experimental sensors equivalent to 25 percent of the baseline scientific instrument complement. Table 3.1-3 illustrates this capability to support a growth in the basic instrument complement. For purposes of determining growth capability, the basic instrument complement consists of the Scatterometer, Altimeter, Large Antenna Multichannel Microwave Radiometer, Coastal Zone Color Scanner, Laser Retroflector Assembly, and Interface Boxes. The CZCS data requirements are listed separately and are not included in determining data-rate and stored-data growth. However, housekeeping data requirements are included in determining data-rate and stored-data growth capability.

Table 3.1-3
Instrument Support Summary

		Basic Instrument Complement	Growth	Total
Weight, kgm		803	200	1003
Power, W		858	215	1073
Data Rate	ALT, SCAT, LAMMR, HK, kbps	96	32	128
	CZCS, Mbps	1.2	0	1.2
Stored Data Bits	ALT, SCAT, LAMMR, HK	5.12×10^8	2.56×10^8	7.68×10^8
	CZCS	1.8×10^9	0	1.8×10^9
Commands		324	81	405

Basic Instrument Complement: ALT, SCAT, LAMMR, CZCS, LRA, 1/F BOXES

Data rate includes housekeeping telemetry.

3.2 SCATTEROMETER (SCAT)

3.2.1 OBJECTIVES

The NOSS Scatterometer (SCAT) is an active microwave radar at 14.6 GHz which provides measurements of radar backscatter coefficient from which the synoptic-scale, ocean-surface vector winds are inferred. The physical basis for this technique is the Bragg scattering of microwaves from centimeter-length capillary ocean waves. The strength of the backscatter varies directly with the capilary wave amplitude, which varies directly with wind speed near the sea surface. Moreover, the radar backscatter is anisotropic; therefore, wind direction can be derived from SCAT measurements at different azimuths.

The specific purpose of the SCAT is to obtain scattering coefficient measurements over the sea and from these deduce the magnitude and direction of the synoptic scale vector wind and the sea surface wind frictional velocity. The SCAT provides surface wind velocity within 2 m/s over a range of 4 to 24 m/s and direction within $\pm 20^{\circ}$.

3.2.2 INSTRUMENT DESCRIPTION

The SCAT, shown in Figure 3.2-1, incorporates six dual linearly-polarized fan-beam antennas, which produce a starlike pattern of illumination on the Earth as shown in Figure 3.2-2. Forward-, side-, and aft-looking antennas are used to obtain three independent radar measurements at three azimuth angles for each resolution cell. The peak of the antenna beam is centered at 47° incidence angle to favor the outer swath section where the received signals are weaker due to increased range and lower scattering coefficient.

The time between illumination of a given resolution cell by the three beams is a function of the cell's position along the fan beam illumination. Each 50-km resolution cell has three footprints (not at the same time, however) giving scattering coefficient data at azimuth angles 45° apart. Three measurements of radar backscatter from each resolution cell and cross polarization measurements help remove wind velocity and direction ambiguities.

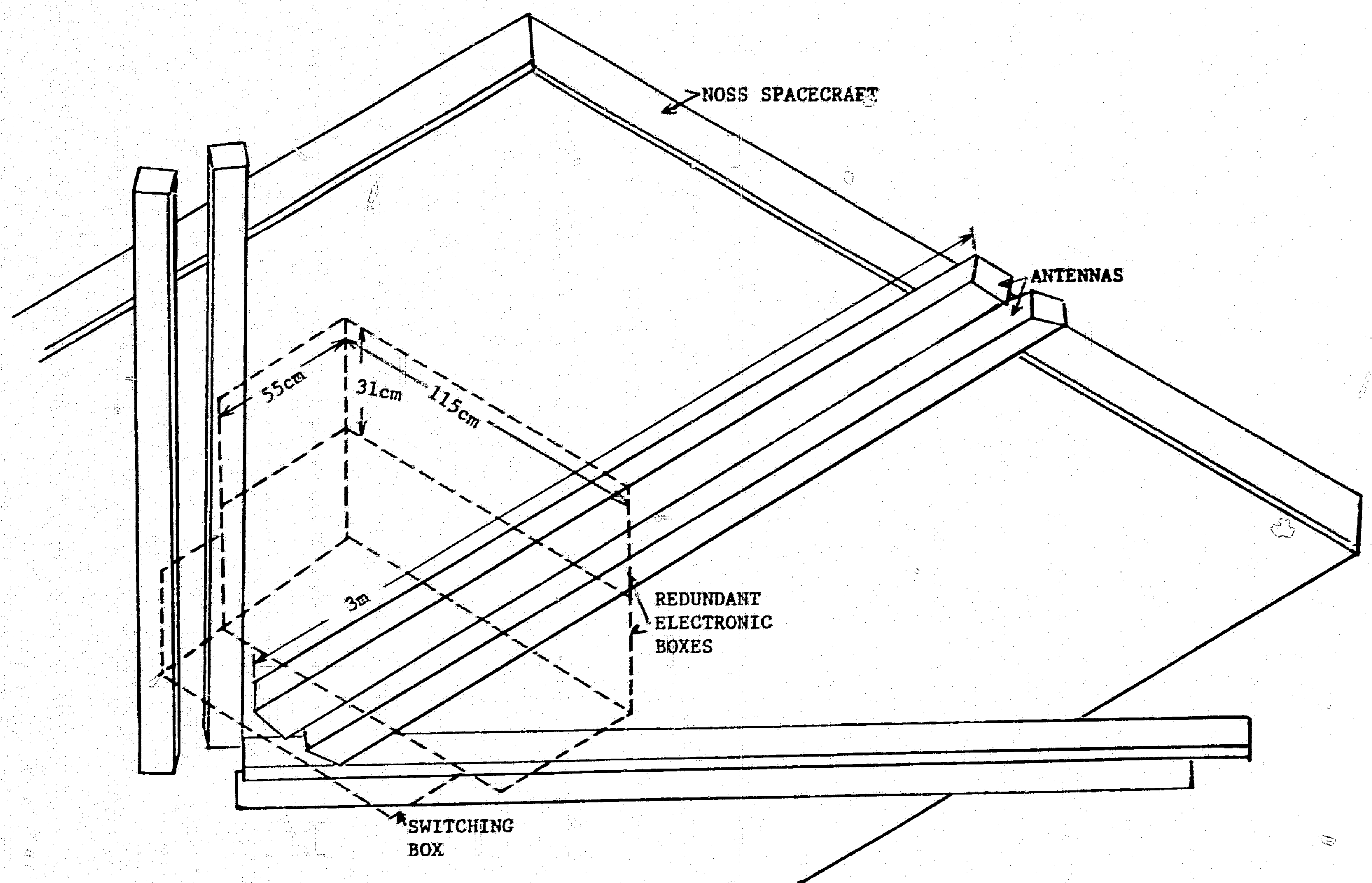


Figure 3.2-1. SCATTEROMETER MAJOR ELEMENTS

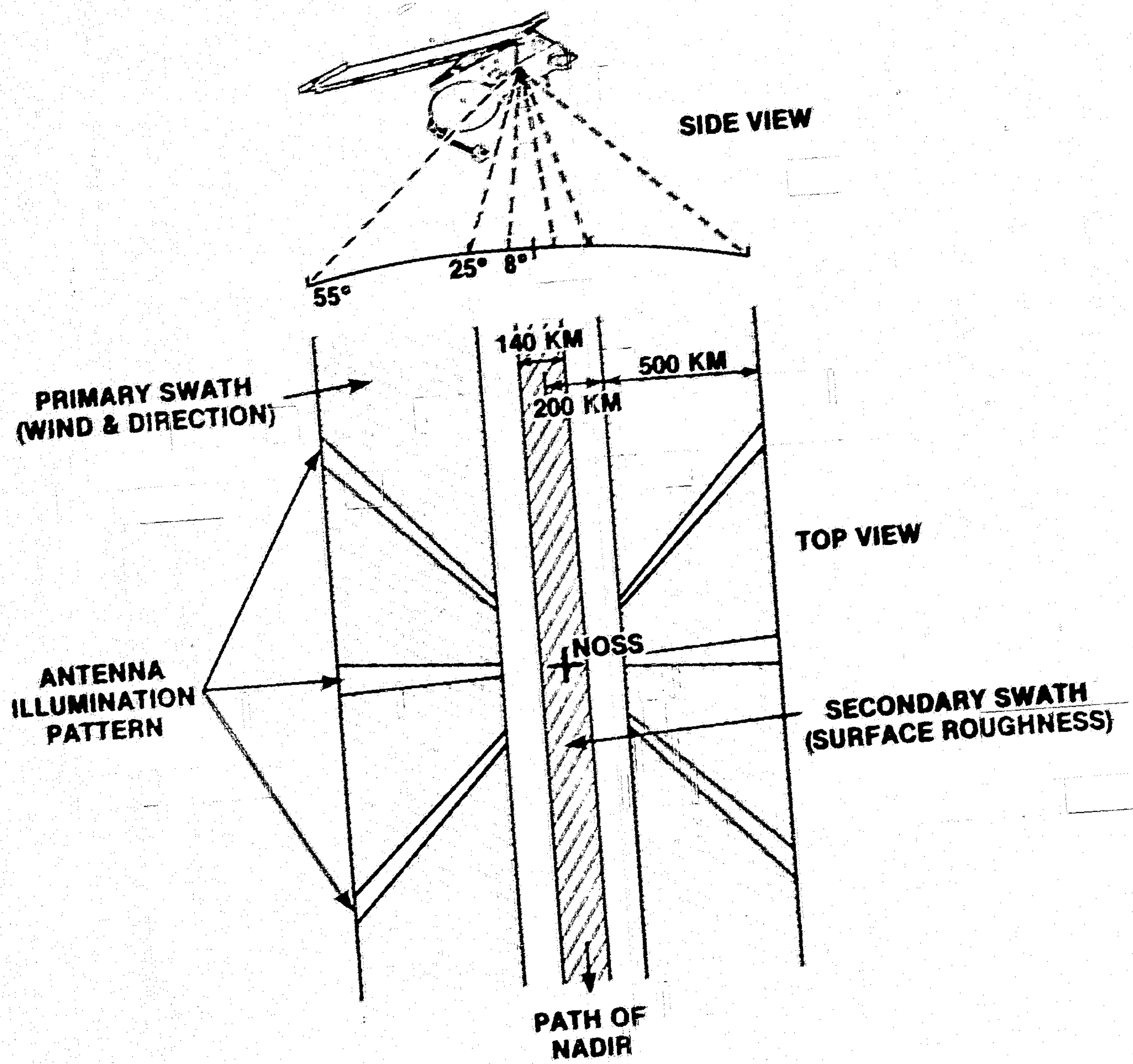


Figure 3.2-2. NOSS Scatterometer Swath Characteristics

A block diagram of the SCAT is shown in Figure 3.2-3. A 100W peak power pulse is directed to the antennas, each of which has two ports, one for each polarization. The 14.6 GHz signal is switched sequentially through six antenna-polarization combinations taking 1.89 seconds each for a total 11.34 seconds to complete one switching sequence. During each 1.89s measurement period, 61 return pulses each about 6 ms long, are processed which results in a 360 ms integration time for the radar return signal.

The return signals backscattered from the ocean surface are fed to a processor which contains 15 processing channels. Fifteen Doppler filters and range gates are used to electronically subdivide the fan beam into separate resolution cells and to obtain the mean ocean scattering coefficient. Three of these channels provide scattering coefficient measurements from incidence angles near nadir from which wind speed in the 140-km swath is inferred. The other 12 channels subdivide the 500-km swath on each side of the NOSS track into resolution cells approximately 50 km on a side in which wind speed and direction are determined. Instrument characteristics are summarized in Table 3.2-1.

Table 3.2-1
Scatterometer Characteristics

Total Weight (kgm)	180
Electronics (2) (kgm),each	60
Antennas (6) (kgm),each	10
Dimensions	
Electronics (cm,each)	115 x 55 x 31
Antennas (cm, each)	300 x 10 x 15
Power (W), average	90 Regulated, 28 ± .3V 75 Unregulated, 28 ± 4V
Peak	400 (unregulated)
Radar Signal Peak Pulse (W)	100
Beam Width (^{o})	½ x 25
Swath Width (km)	1400
Resolution Cell (km)	50 x 50
Data Rate (kbps)	4

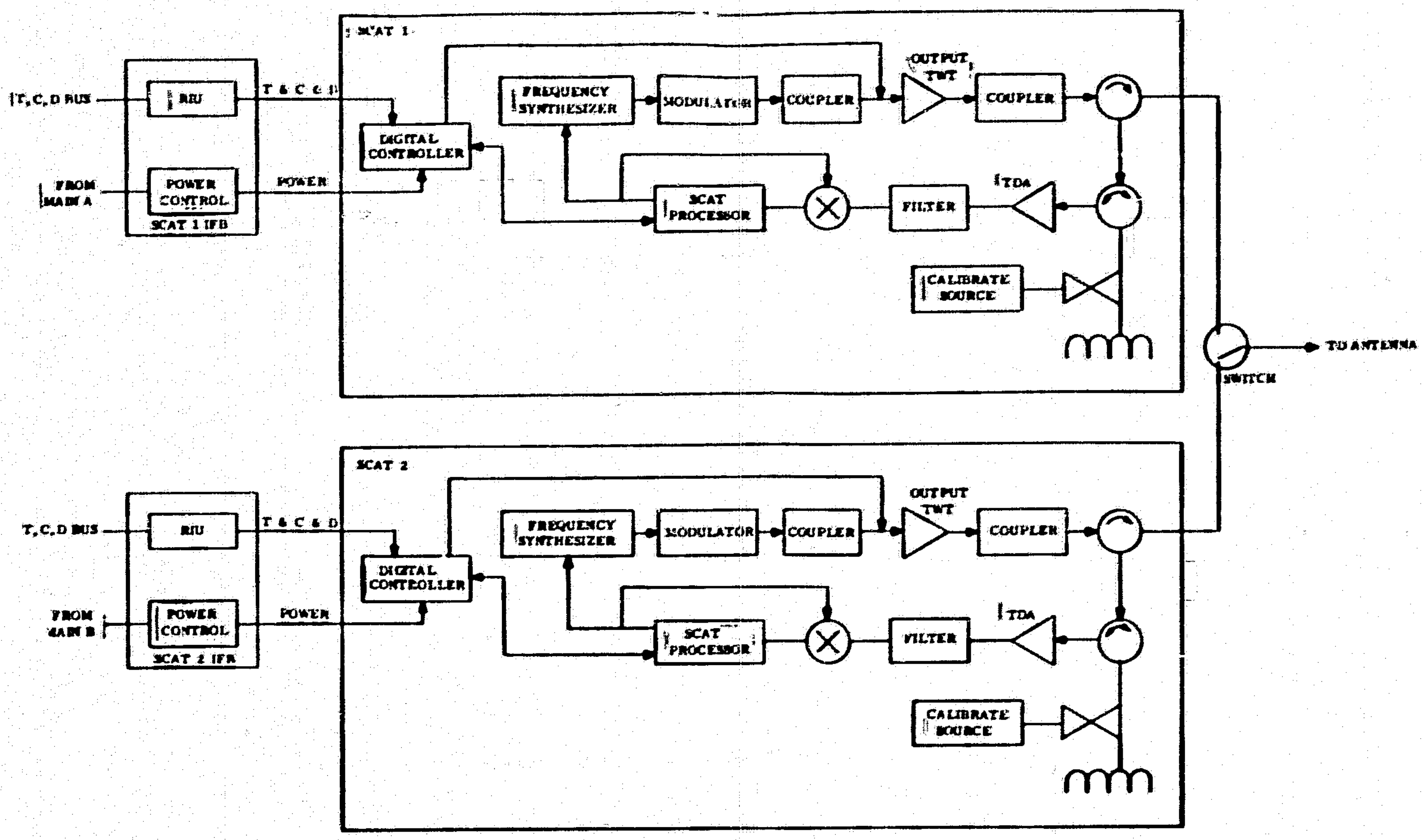


Figure 3.2-3. Scatterometer Block Diagram

3.2.3 INTERFACES

A digital controller accepts spacecraft commands and power, generates the precise timing and control logic needed by the scatterometer to form rf pulses, and operates the processor. In addition, it accepts the output scatterometer data and instrument housekeeping parameters and formats them so they are compatible with the spacecraft data systems.

Thirty total commands are sent to the scatterometer. Six commands are latching relays with relay coils from 125 ohm to 15 kohm resistance. Redundant input is from +20 to +32 volts. Eighteen commands are driven by a relay closure on the spacecraft to discharge a charged capacitor driving L5400 logic in the sensor.

Electrical power is provided to the scatterometer on a 28 Vdc regulated bus. The instrument utilizes a 5 MHz ($\pm$0.5 Hz) reference signal from the spacecraft to generate all timing and radio frequency signals.

Instrument electronics cannot be subjected to a field of greater than five gauss when measured at the SCAT package. Ferro-megnetic materials cannot be closer than 15.3 cm (6 inches) to the electronics package for longer than thirty minutes.

Scatterometer telemetry consists of bi-level status, bi-level fault determination, and analog housekeeping (instrument health) data. Bi-level status data includes synch pattern, gain bits, SLO frequency selected, calibration status, circulator status, mode selected and analog temperature monitor subcommutator identification bits. Bi-level fault determination data includes input current, body current and undervoltage trips, receiver protect circulator status, and phaselock loop indicators. Analog instrument health data includes internal voltages, currents, rf powers, component and baseplate temperatures, and 40 SCAT antenna temperatures.

3.3 ALTIMETER (ALT)

3.3.1 OBJECTIVE

The altimeter will provide all-weather, global monitoring of the ocean wave height and departures of the sea surface from the marine geoid corresponding to ocean dynamic processes. Control of the sensor operation modes provide a capability of measuring significant wave heights at high sea states (to 25 meters) along the sub-satellite track.

3.3.2 INSTRUMENT DESCRIPTION

The basic measurement process involves transmitting a pulse and analyzing the leading edge of the return pulse. The round-trip delay time gives the overall range for determining the large scale marine topography. The return pulse leading edge is also "smeared" due to different path lengths from the peak and troughs of the waves in the field-of-view and the processing of this "smearing" yields the significant wave height. This is depicted in Figure 3.3-1.

This instrument is an active, nadir looking radar which operates at a nominal frequency of 13.5 GHz. A linear chirp pulse of 320 MHz bandwidth and 3.2 μs duration is transmitted through a 1 m diameter antenna. Figure 3.3-2 is a simplified block diagram of the instrument showing both the rf and the digital portions.

The instrument uses active pulse compression to achieve the fine resolution necessary for precision height tracking. Full-deramp processing is used to generate a transmit chirped pulse and to de-chirp the received signals. A single filter (expander) is retriggered to produce a properly timed chirped local oscillator pulse that brackets the received signal.

The full-deramp process compresses the bandwidth of the received signals of interest to $\pm$10 MHz even though the transmitted pulse has a 320 MHz bandwidth. A fully digital filter bank uses medium speed devices at a maximum clocking speed of 20 MHz. Since the pulse rate is 1020 pulses per second and all

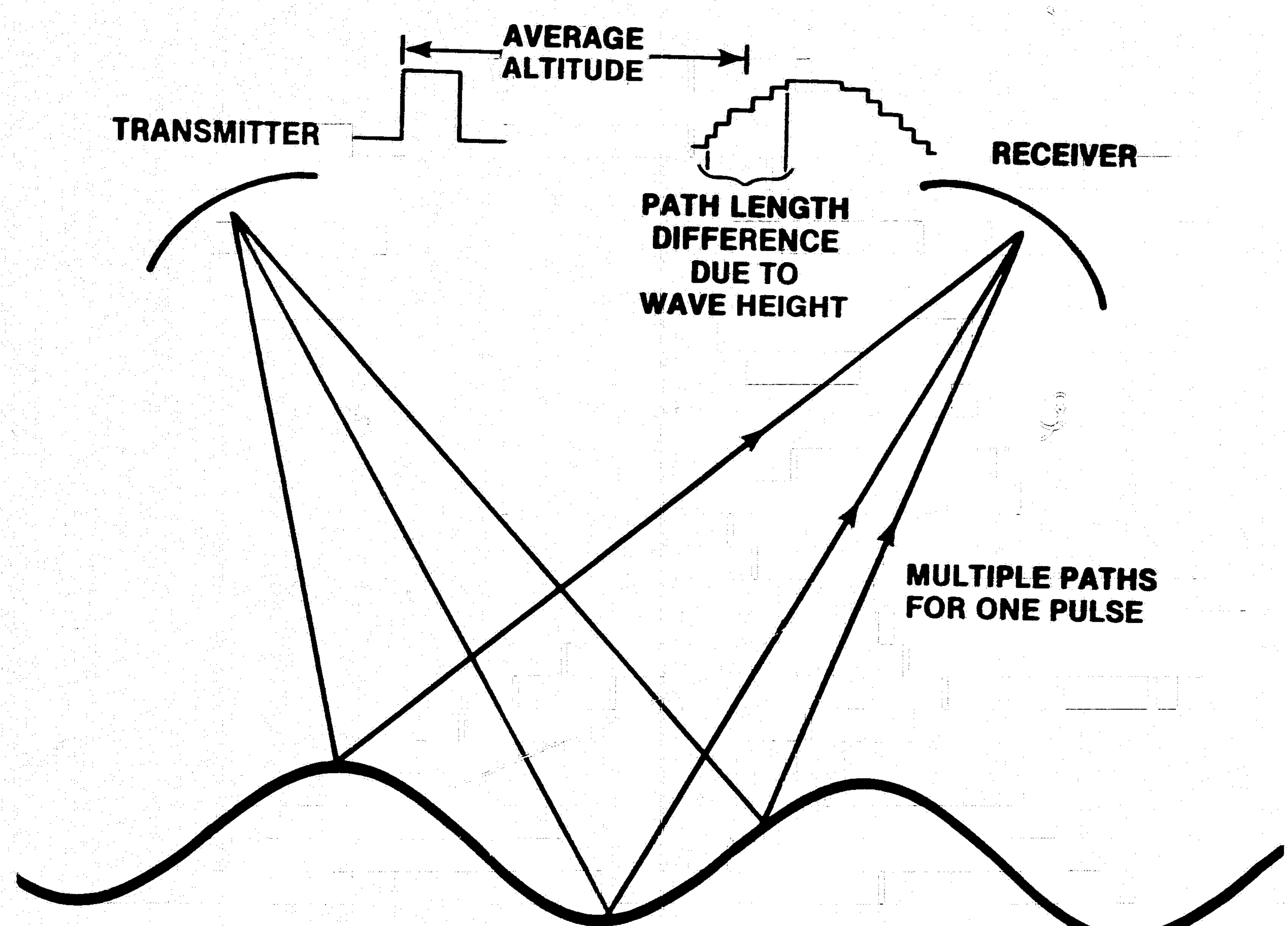


Figure 3.3-1 Simplified Altimeter Operation

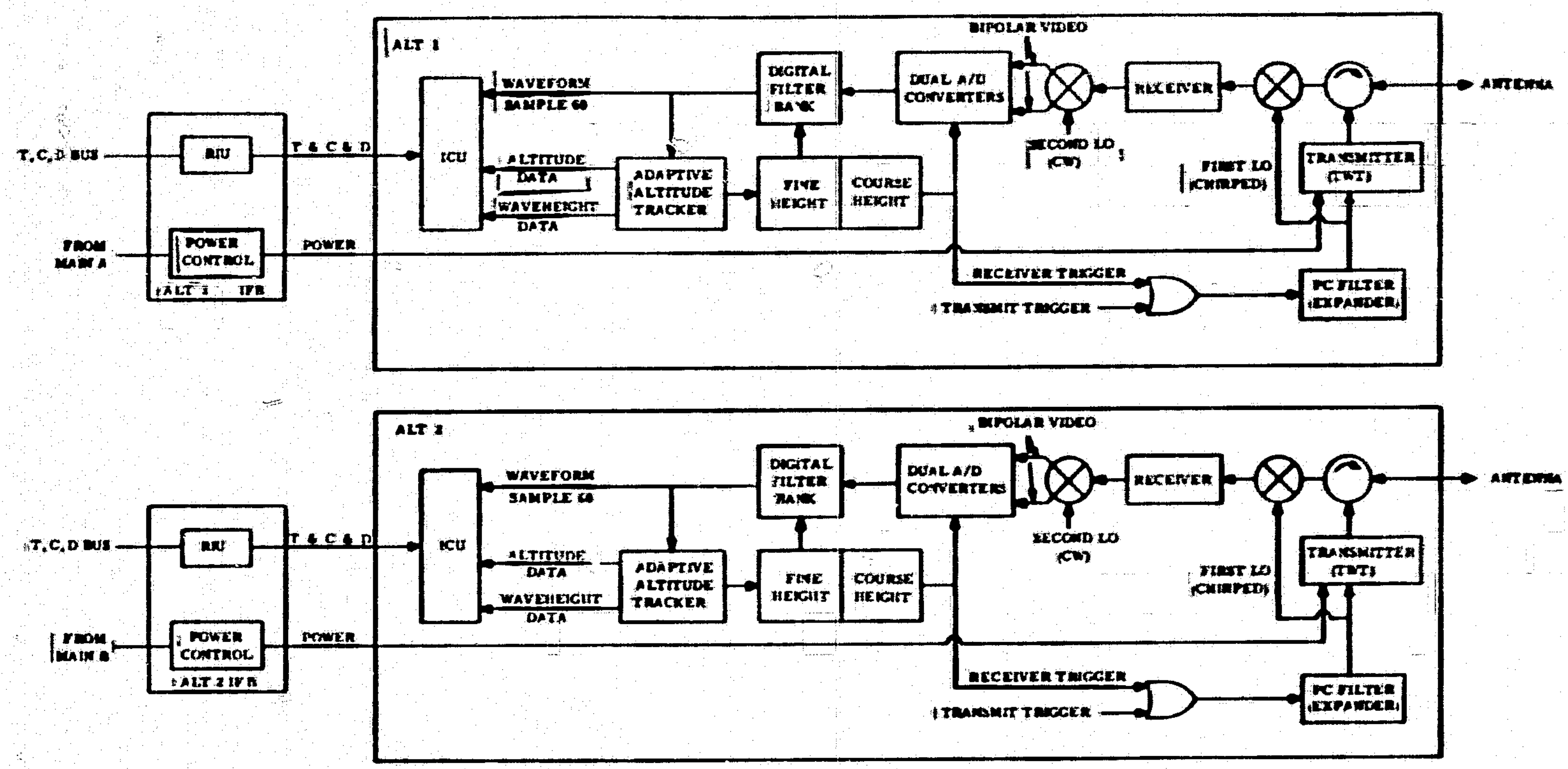


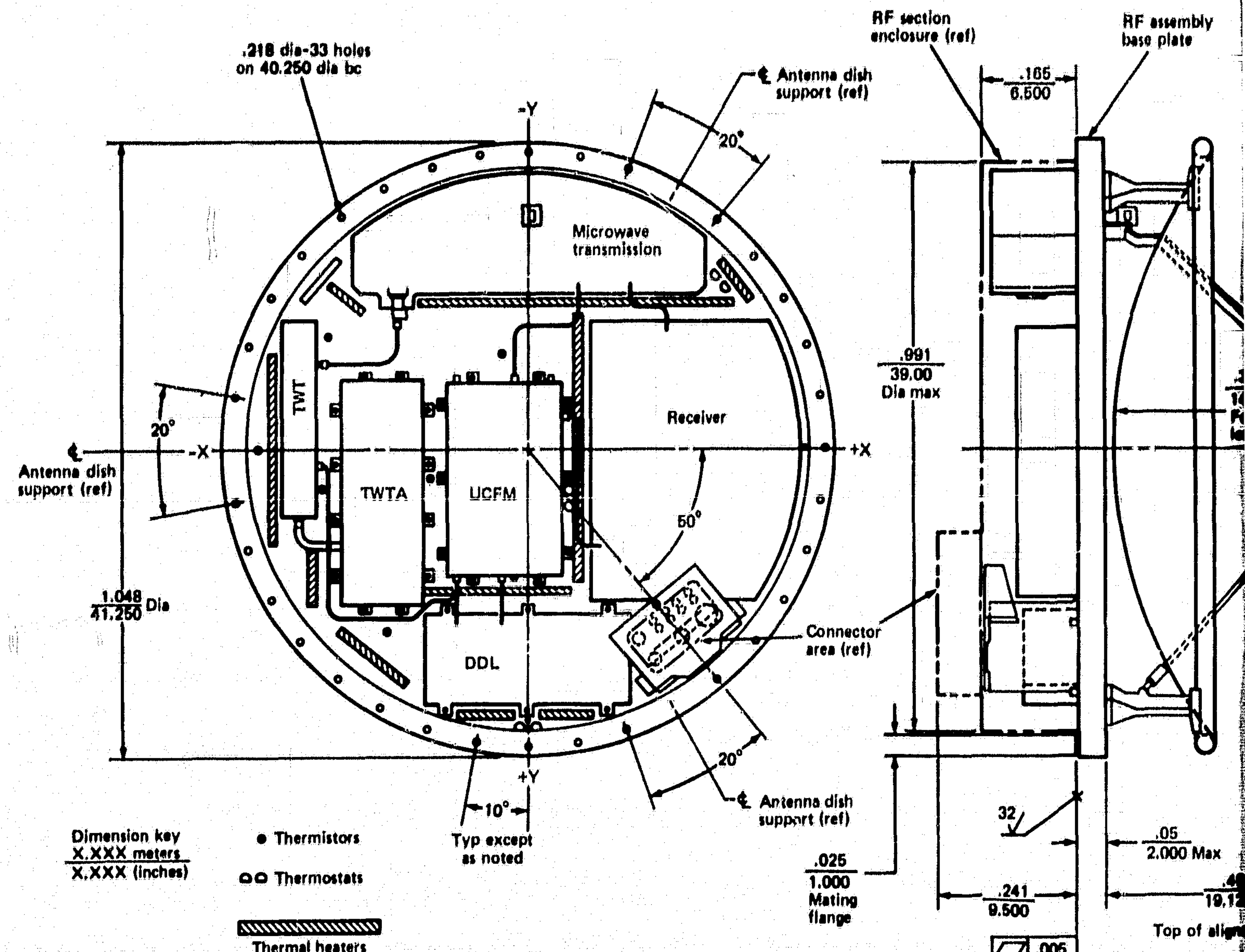
Figure 3.3-2. ALTIMETER SYSTEMS BLOCK DIAGRAM

received signal information is confined to the 3.2 μs pulsed local oscillator interval and during that interval, signal returns are A/D converted and stored, the remaining 980 μs are available for processing. Rather than direct implementation of 60 filters (full deramp transforms time resolution to frequency resolution), the same digitized data are accessed 60 times sequentially and in effect one filter is used 60 times with appropriately offset center frequencies to produce the desired 312.5 kHz (3.125 ns) spacing. At the same time the entire filter bank is moved in 5 kHz (0.05 ns) steps for fine height tracking.

A microcomputer-based adaptive tracker unit processes information contained in the waveform samples to yield a significant waveheight(the average of the largest third of all ocean waves, H-1/3) estimate and this,in turn, is used to adjust tracking loop parameters to compensate for the effects of H-1/3 variation up to 20 meters. The adaptive tracker also decodes commands and sequences the altimeter through acquisition, track, and calibration states; and formats science and engineering data for TM output.

The two primary outputs of the radar altimeter are the precision height from which the ocean geoid and surface topography are determined and the waveheight (H-1/3) estimate computed on-board in the adaptive tracker. In addition, receiver gain control settings (AGC in dB) are provided in both operate and calibrate modes from which the surface reflectivity may be determined in subsequent data processing. The altimeter is designed for continuous operation in support of the global coverage requirement of the mission. Figures 3.3-3 and 3.3-4 show the two major elements of the system, the rf section and the signal processors. The parabolic dish antenna is attached to a 5.1-cm (2-inch) thick circular honeycomb deck and on the opposite face is the rf section. The signal processor section is mounted on a rectangular plate and both sections have covers for RFI/EMI shielding.

The 5000-hour operating life predicted for the traveling wave tube (TWT) is based on defining end-of-life as a 1 dB drop in peak power output capability. There is ample margin in the overall link equation for at least a 3 dB drop and, insofar as the gradual decrease in power due to cathode



FOLDOUT FRAME

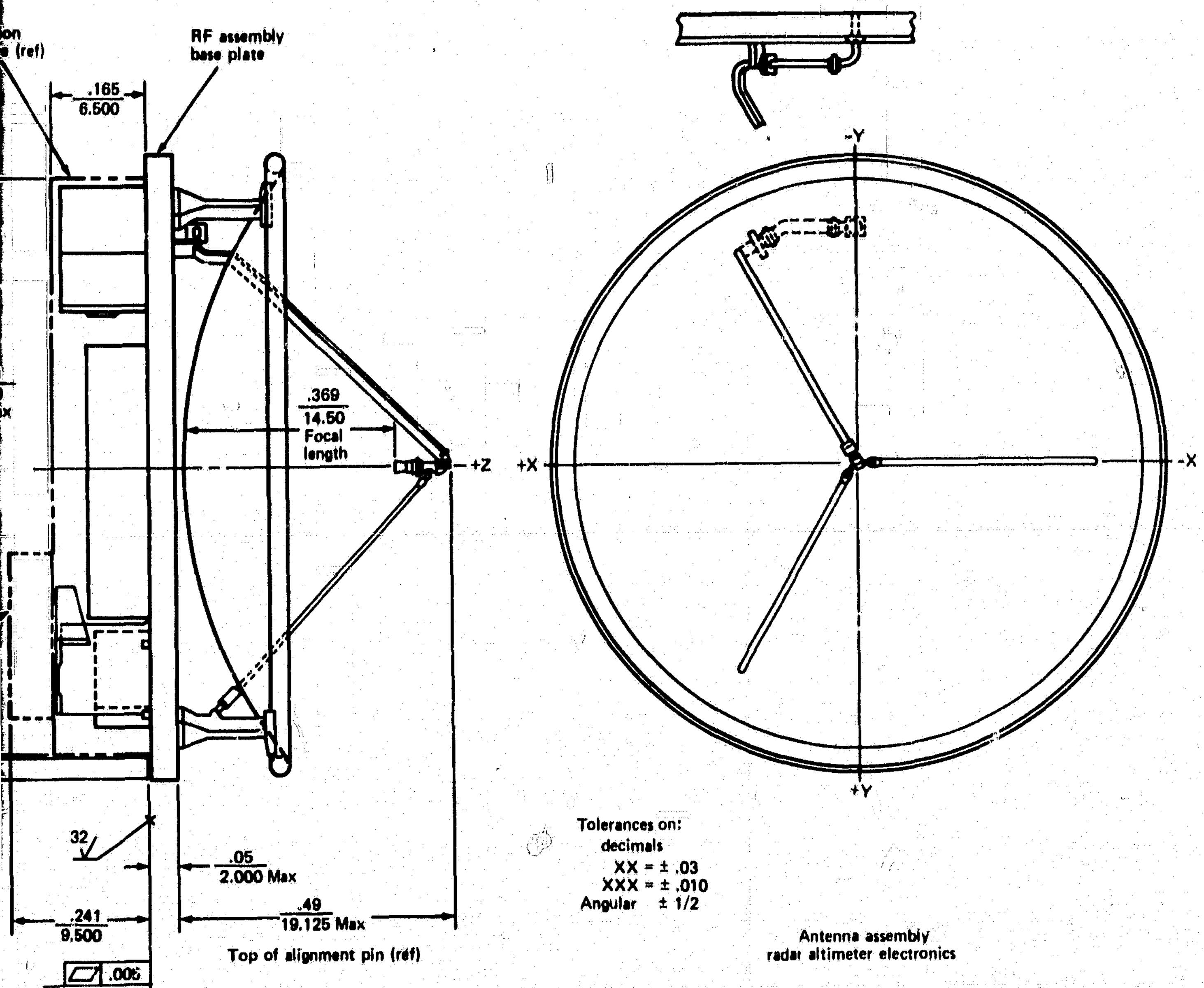


Figure 3.3-3. Radar Altimeter rf Section

FOLDOUT FRAME 2

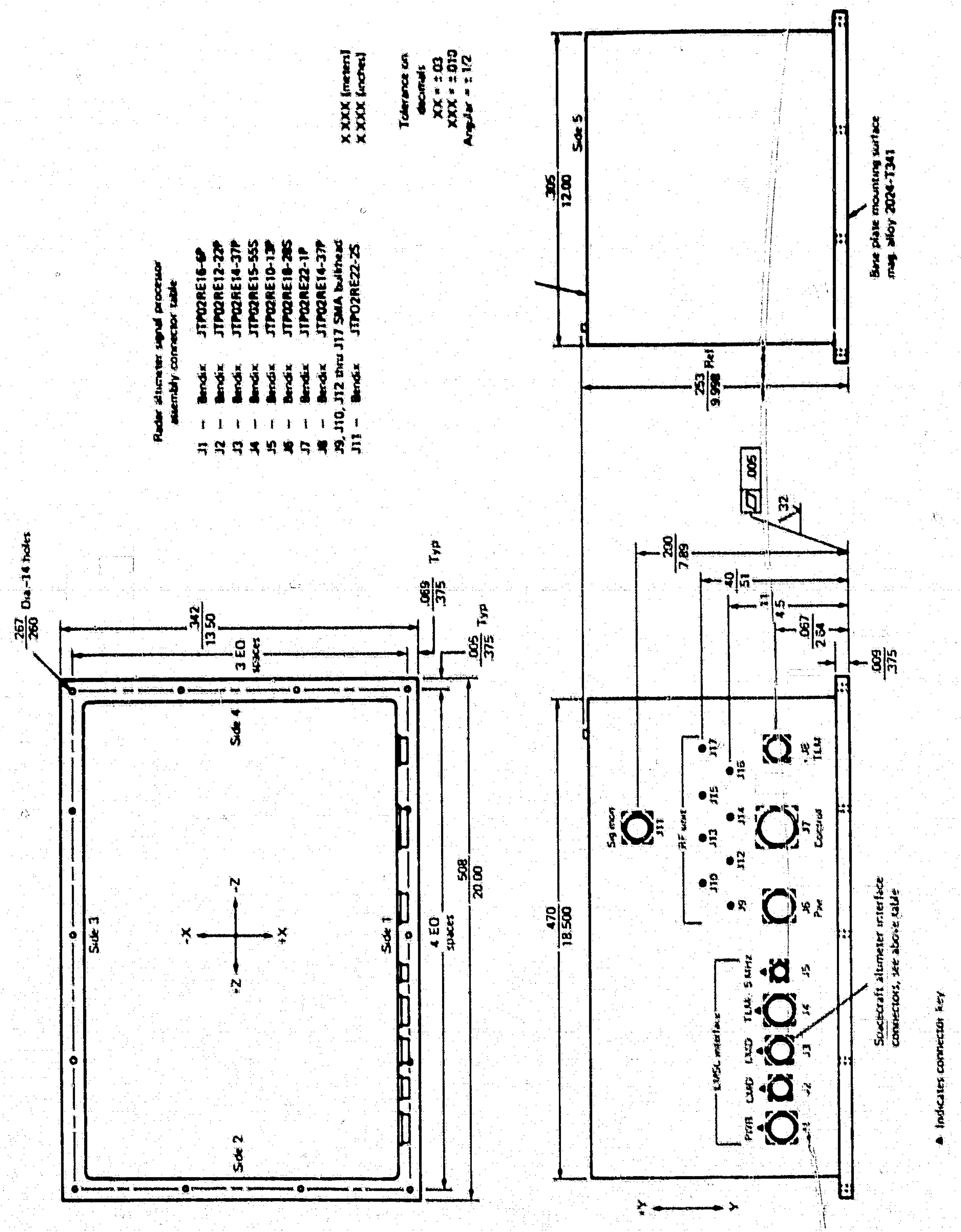


Figure 3.3-4. Radar Altimeter Signal Processor

depletion is the only mechanism limiting lifetime, continuous operation for 1 year (8760 hours) should present no problem. Complete instrument redundancy (i.e., two complete altimeters) are being flown on NOSS to provide for a 3-year lifetime.

The antenna is a horn-fed parabolic dish 1 m in diameter with a gain of 40.8 dB. A unique feature of the antenna is the independent attachment of the feed and its supports and the spun aluminum dish and its supports.

The microwave transmission unit provides the waveguide interconnections between the antenna, transmitter, and receiver.

The traveling wave tube amplifier (TWTA) consists of a high voltage power supply/modulator and an external TWT joined by an external splice block. The TWT operates at a cathode voltage of -11 kV and derives its prime power directly from the spacecraft 28V bus, but also makes limited use of the regulated voltages from the low voltage power supply.

During the normal track mode, the altimeter uses a 3.2 μs chirped local oscillator pulse and can only process returns within this window. The acquisition range for the altimeter is 761 to 868 km and the acquisition procedure consists of a 0.2s cw phase, a 2s (minimum) chirp phase with short time constant height and AGC tracking, followed by chirp tracking with nominal loop time constants. Figure 3.3-5 shows the NOSS altimeter scan parameter geometry.

Based on waveform sample data averaged for 50 radar pulse time intervals (PRT), the adaptive tracker processes the samples to form gates, estimate waveheight, compute a height error, and update the previous height and height rate estimates in the following 50-PRT interval. Table 3.3-1 is a listing of the main parameters of the altimeter.

3.3.3 INTERFACES

All electrical interfaces to the spacecraft (except for the rf section heater/thermostats) are made to the signal processor through a connector panel and filter.

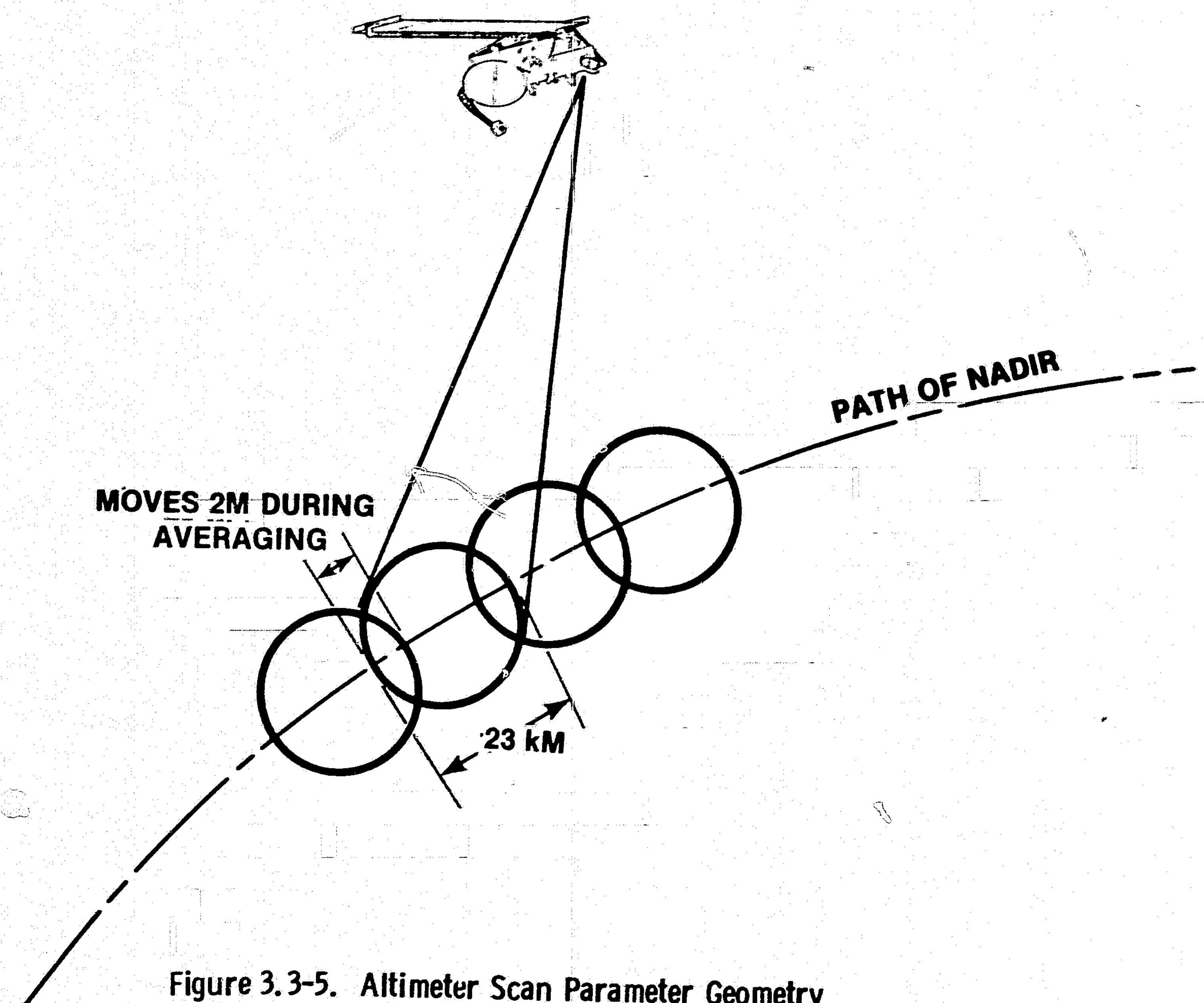


Figure 3.3-5. Altimeter Scan Parameter Geometry

Table 3.3-1

Altimeter Characteristics

Antenna beamwidth (^{o})	1.6
Frequency (GHz)	13.5
Peak rf power (kW)	2
Average rf power (W)	6.5
Pulse width (uncompressed) μs	3.2
Pulse width (compressed) ns	3.125
Pulse repetition rate (Hz)	1020
Footprint diameter (km)	23
Altitude precison (cm)	<10
Wave Height Precision (m)	$\approx$0.5 below 5m, 10% above 5m to a maximum of 19.9m
Antenna Gain (dB)	40.8
Weight (kg, each)	100
Power (W, each)	102 Standby 177 In Use
Dimensions (cm, each)	
Signal Processor	34 x 51 x 25
Antenna and rf	100 x 80

Two types of commands are sent to the altimeter. Pulse commands activate latching relays and data commands are sent by a 10-bit parallel binary word latched into the altimeters via an instrument-unique strobe pulse.

The altimeter outputs 85 10-bit science data words every 98 ms (100 PRTs). The data are output in serial form along with a clock and a burst gate that lasts 850 clock periods. A subcommutated engineering data word provides up to 46 measurements (not all used) at a 4.6s sampling interval. In addition, 12 analog health monitor functions are provided as a separate telemetry output.

Electrical power is provided to the altimeter on a 28V (nominal) bus. Power consumption for various operating modes shows that the rise in power vs. bus voltage in the standby mode results from a current limiting auxiliary regulator in the TWTA, supplying mainly heater power to the TWT. When high voltage is switched ON, this regulator is bypassed.

The altimeter requires a 5 MHz reference signal from the spacecraft Doppler beacon to generate all timing and radio frequency signals. The reference frequency is offset -50 ppm (250 Hz) from the nominal 5 MHz. This offset must be included in the conversion from two-way height delay to height in meters, since it represents a 40m difference at 800 km.

Temperature control of the rf section is accomplished by heat flow through the mounting ring surface to a spacecraft radiator. Temperature control of the signal processor is accomplished by heat flow through the baseplate to a spacecraft radiator as well as by radiation from three silver Teflon-coated side panels.

A Laser Retroflector Array (LRA) is mounted toroidially about one altimeter antenna and is used with ground laser tracking to provide post-facto precise calibration.

3.4 LARGE ANTENNA MULTICHANNEL MICROWAVE RADIOMETER (LAMMR)

The Large Antenna Multichannel Microwave Radiometer (LAMMR) is a mechanically-scanned, multifrequency passive microwave sensor that in its primary form consists of five dual channel , dual polarization, total power radiometers at five microwave frequency bands between 4.3 and 36.5 GHz using a single high-gain antenna. The 3.6m offset paraboloid reflector and multichannel feed system mechanically scans at 1 rps and maps the Earth's surface and atmosphere by obtaining high resolution radiometric measurements. The brightness temperatures are obtained at discrete microwave frequencies resulting in the ability to infer geophysical properties such as sea surface temperature, oceanic wind speed, precipitation, water vapor, sea ice concentration, and snow properties.

The LAMMR scans at 1 rps by rotation of the entire reflector, feed, and receiver assembly. The LAMMR is mounted on the instrument platform on the Earth side of the spacecraft to clear all portions of the spacecraft with the main beam. The scan is obtained by continuously rotating the beam approximately 43° off the nadir axis resulting in a 1300-km surface swath. The LAMMR spatial resolution for the primary channels varies with the frequency and ranges between 7 and 40 km.

3.4.1 OBJECTIVE

The primary purpose of the LAMMR is to obtain ocean surface and atmospheric parameters on a nearly all-weather operational basis. The LAMMR simultaneously measures microwave thermal emission from the Earth's atmosphere and surface in five channels of different center frequency and wavelength. These microwave brightness temperatures are used, in turn, to estimate such parameters as sea surface temperature, sea ice concentrations, and wind speed.

3.4.2 INSTRUMENT DESCRIPTION

The LAMMR embodies a 3.6m aperature antenna which scans the Earth by rotating through 360° about the spacecraft nadir with a 43° (off-nadir) half cone angle. Data sampling occurs during the forward 120° of

scan as shown in Figure 3.4-1. The instrument is calibrated during a 60°-segment on each side of the data sampling scan. The rear 120° is not used for radiometry. The 1300-km swath width is developed by orienting the beam at a 43° angle to the satellite nadir. Momentum compensation is provided to minimize unbalanced momentum inputs to the spacecraft attitude control system. A mechanical rotational joint with slip rings provides bearing and lubrication to insure long-life operation and power and signal transfer across the joint. The complete sensor (reflector, feeds, and receivers) is rotated relative to the spacecraft platform. All feeds and electronics are mounted at the focal point of the main reflector to minimize any losses.

The primary LAMMR instrument simultaneously collects microwave thermal emission from the Earth's surface and atmosphere in five channels of different center frequency, each of which measures radiation in two orthogonal linear polarizations. The feed system is connected to the five dual radiometer inputs either through an orthomode transducer or directly for separating the polarizations. Calibration will be achieved by switching to sky horns during the calibration segment of the scan. A simple block diagram is shown in Figure 3.4-2.

The radiometers detect, amplify, and convert the signals to digital data which are multiplexed with instrument engineering and calibration data and fed to the satellite data system for transmission to the ground.

The microwave brightness temperatures of the ocean surface are corrected for interfering surface effects as well as atmospheric absorption and emission. Measurements from the 10 channels (5 frequencies and dual polarizations) are analyzed to decouple the interaction of several sea surface and atmospheric phenomena which affect the measured brightness temperatures. Wind speed information is contained primarily in the horizontal channels of all wavelengths. Atmospheric parameters are not generally polarized, but have distinctive spectral signatures. Significant instrument characteristics are listed in Table 3.4-1.

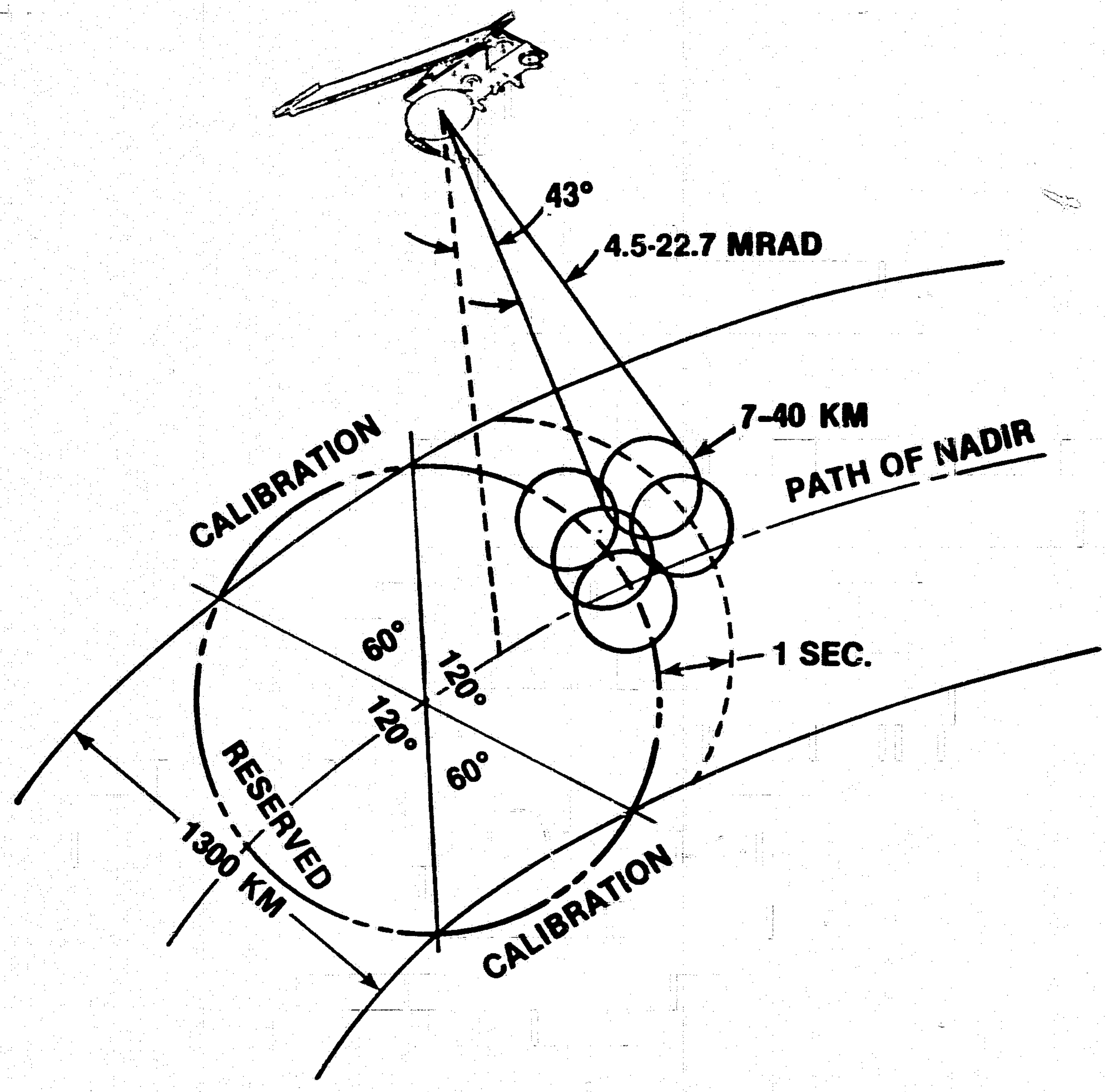


Figure 3.4-1. LAMMR Scan Parameter Geometry

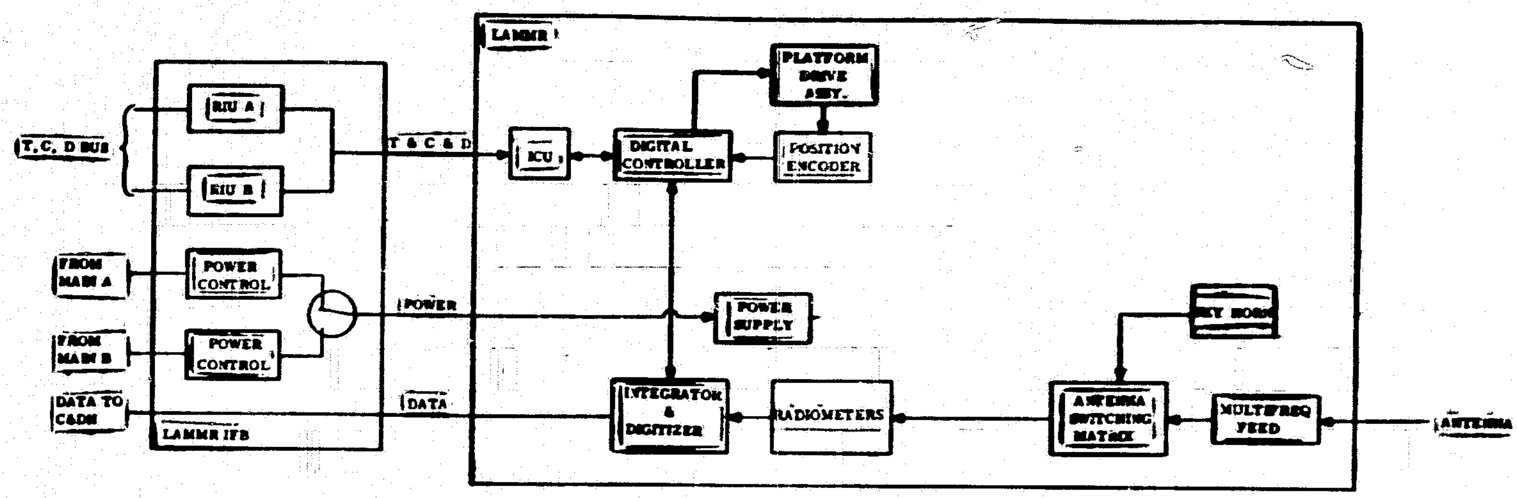


Figure 3.4-2. LAMMR BLOCK DIAGRAM

TABLE 3.4-1

SIGNIFICANT LAMMR CHARACTERISTICS

Weight (kgm)	320
Effective Aperature (m)	3.6
Swept Volume in Orbit (m^3)	140
Power (W)	350
Primary Radiometer Frequencies (GHz)	4.3, 10.65, 18.7, 21.3, 36.5
Primary Radiometer Wavelengths (cm)	0.8, 1.4, 1.6, 2.8, 6.9
Scan Rate (rps)	1
Swath Width (km) (for 120° Scan)	1300
Beamwidth (°)	0.26 to 1.3
Footprint Diameter (km) (Primary channels)	7 to 40
Water Temperature Precision (°K)	1
Wind Speed Precision (m/s)	2
Pointing Knowledge (°)	0.03 (3 σ)

LAMMR data can distinguish between open water and sea ice in partial beam-filling situations because of the different spectral signatures of these two targets. Locating the position and extent of open areas in the polar sea ice cover is very important for obtaining the heat balance in the polar regions. First and multiyear sea ice can also be distinguished by microwave brightness temperature signatures. The motion of multiyear ice within the canopy can be used to study ocean circulation and ice dynamics in polar regions.

3.4.3 INTERFACES

The spacecraft provides +28 ± 0.3 Vdc regulated and +28 Vdc (nominal) unregulated electrical power to the instrument.

Two types of commands are sent to the LAMMR. Pulse commands (+20 to +32 V pulse) activate 30 relay coils and 8 data commands consist of a +4.5 ± 1.0V pulse. The LAMMR telemetry output consists of 35 analog and 100 digital functions.

Uncompensated momentum is less than one foot-pound second. Knowledge of instrument pointing position within 0.03° is required.

3.5 COASTAL ZONE COLOR SCANNER (CZCS)

3.5.1 MISSION OBJECTIVES

The purpose of the Coastal Zone Color Scanner (CZCS) is to measure the concentration of chlorophyll near the sea surface. This indicates the abundance of phytoplankton (planktonic plants) which contain chlorophyll and are at the bottom of the oceanic food chain.

Additional CZCS objectives are the measurement of sediment gelbstoffe (dissolved organic compounds) and sea surface temperature.

3.5.2 INSTRUMENT DESCRIPTION

The multi-spectral imaging Coastal Zone Color Scanner, shown in Figures 3.5-1 and 3.5-2, is an Earth-scanning nine-channel (detector) radiometer using a classical Cassegrainian telescope and a Wadsworth-type grating spectrometer. All nine detectors observe the same area on the Earth's surface at the same time and differ only in the spectral range (or color) that they detect.

The CZCS optical system separates the scan scenes into two spectral ranges, the visible (including solar infrared) and the thermal infrared, by a dichroic beam splitter. The visible light is depolarized and then dispersed by the diffraction grating. Each of the seven wavelengths or colors is sensed by a separate silicon photodiode detector. Co-registration of the detectors is assured by the use of a single, common field stop prior to the spectrometer.

The infrared radiance is directed to two photoconductor detectors mounted to the inner stage of a radiative cooler. Through radiative transfer, the detector is cooled to 120°K (-234°F) with temperature control provided electronically. The door of the radiative cooler is opened during cooler operation and shields the cooler from a direct look at the Earth. Studies are currently underway to modify the baseline six-channel Nimbus-7 CZCS to a nine-channel radiometer by adding two channels in the visible and one in the thermal infrared.

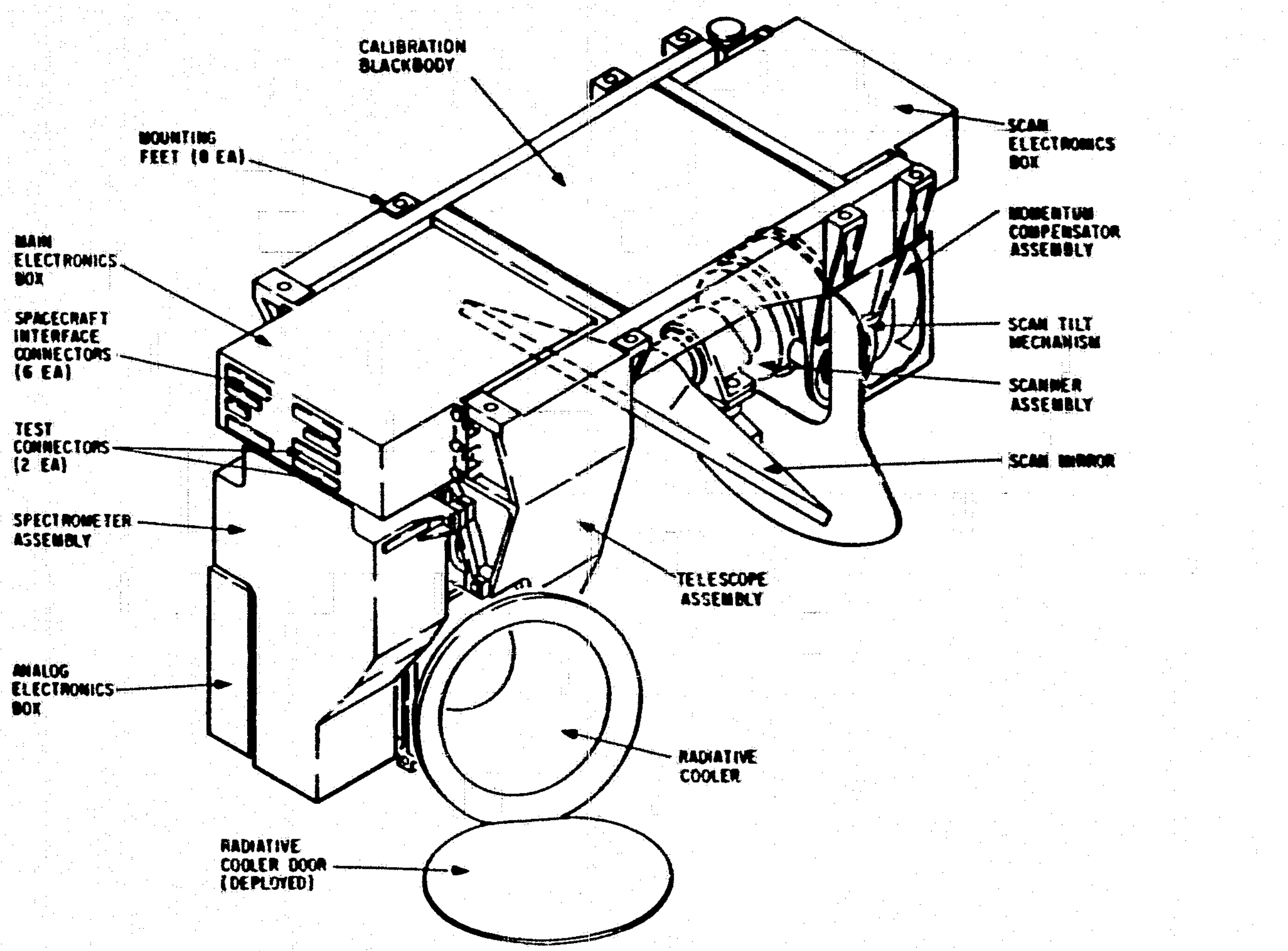


Figure 3.5-1. Major Elements of the CZCS

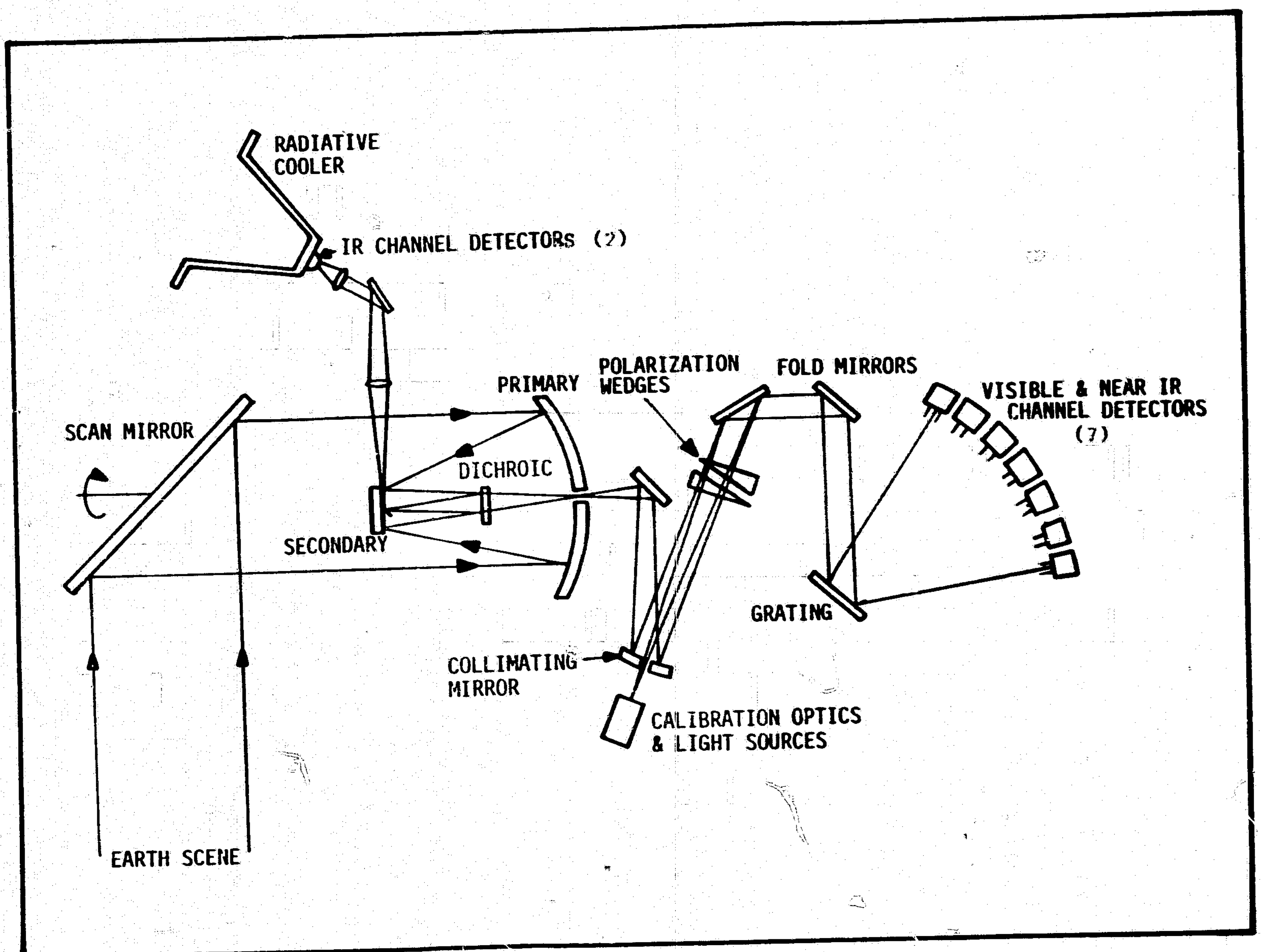


Figure 3.5-2. Optical Schematic

A continuously rotating mirror scans a nominal 0.865 mrad (0.05 degree) Instantaneous Field-Of-View (IFOV) across the Earth's surface perpendicular to the orbit track at 8.08 revolutions per second. The spacecraft's orbital velocity provides the other scan direction. At the orbital altitude of 800 km (500 miles), this results in an instantaneous view of the earth's surface of 0.7 km (0.43 mile) square. An unobstructed scan angle of approximately ± 40 degrees on either side of nadir produces a scan width on the ground of 1340 km (840 mi). Putting it another way, a single scan of the mirror provides a view of the earth 1340 km long and 0.7 km wide. This is shown in Figure 3.5-3. The spacecraft motion is such that successive scans are contiguous , resulting in a complete raster. The rotation of the Earth under the spacecraft allows for total surface coverage in the temperate and polar zones and approximately 80 percent coverage in the tropics everyday.

The instrument has the capability of minimizing solar glint (which can distort the reflected water color) by tilting, upon command, the scan mirror about the telescope axis from its nominal 45 degrees. This results in the instrument field-of-view looking ahead or behind the spacecraft and away from the glint. The field-of-view tilt adjustments can be made in two-degree increments over a ± 20 degree range from nadir.

The scan mirror rotates a full 360° and data sampling occurs throughout the entire scan. As can be seen from Figure 3.5-4, however, only data from selected positions of the scan is required. An entire revolution takes 123.75 ms and is sampled every 13.75 μs. Since there are nine channels and eight bits per sample, a total bit rate of 5.24 Mbps is generated. The data is only used over a percentage (78.72°) of the scan and it appears that the spacecraft should be able to buffer the data rate to about 1.2 Mbps.

A simplified CZCS system/spacecraft interface block diagram is provided in Figure 3.5-5 and a more complex block diagram of the six-channel instrument is shown in Figure 3.5-6. The detailed block diagram is for the Nimbus instrument; the NOSS instrument will be modified to include three additional channels. System characteristics are summarized in Table 3.5-1.

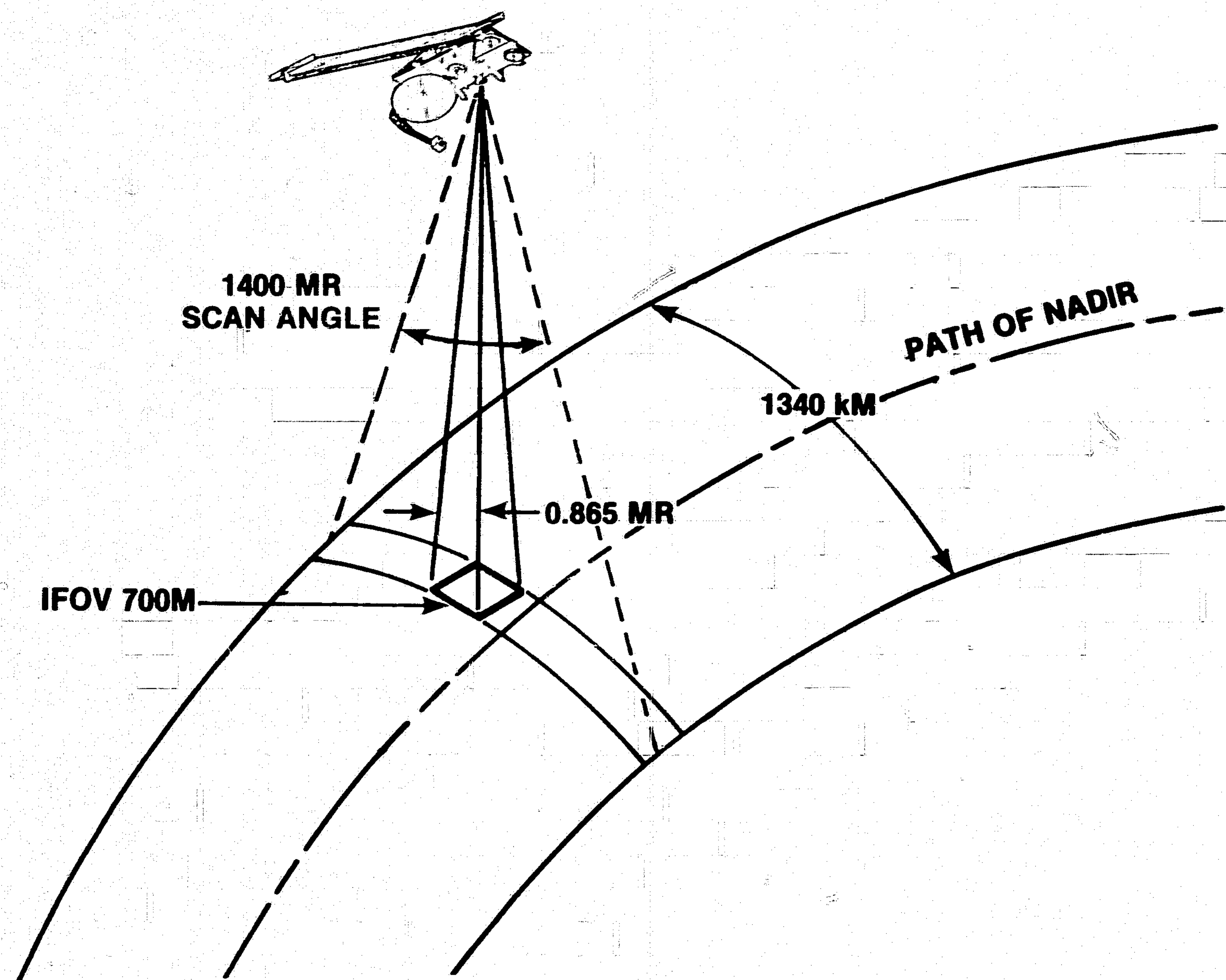


Figure 3.5-3. CZCS Scan Parameter Geometry

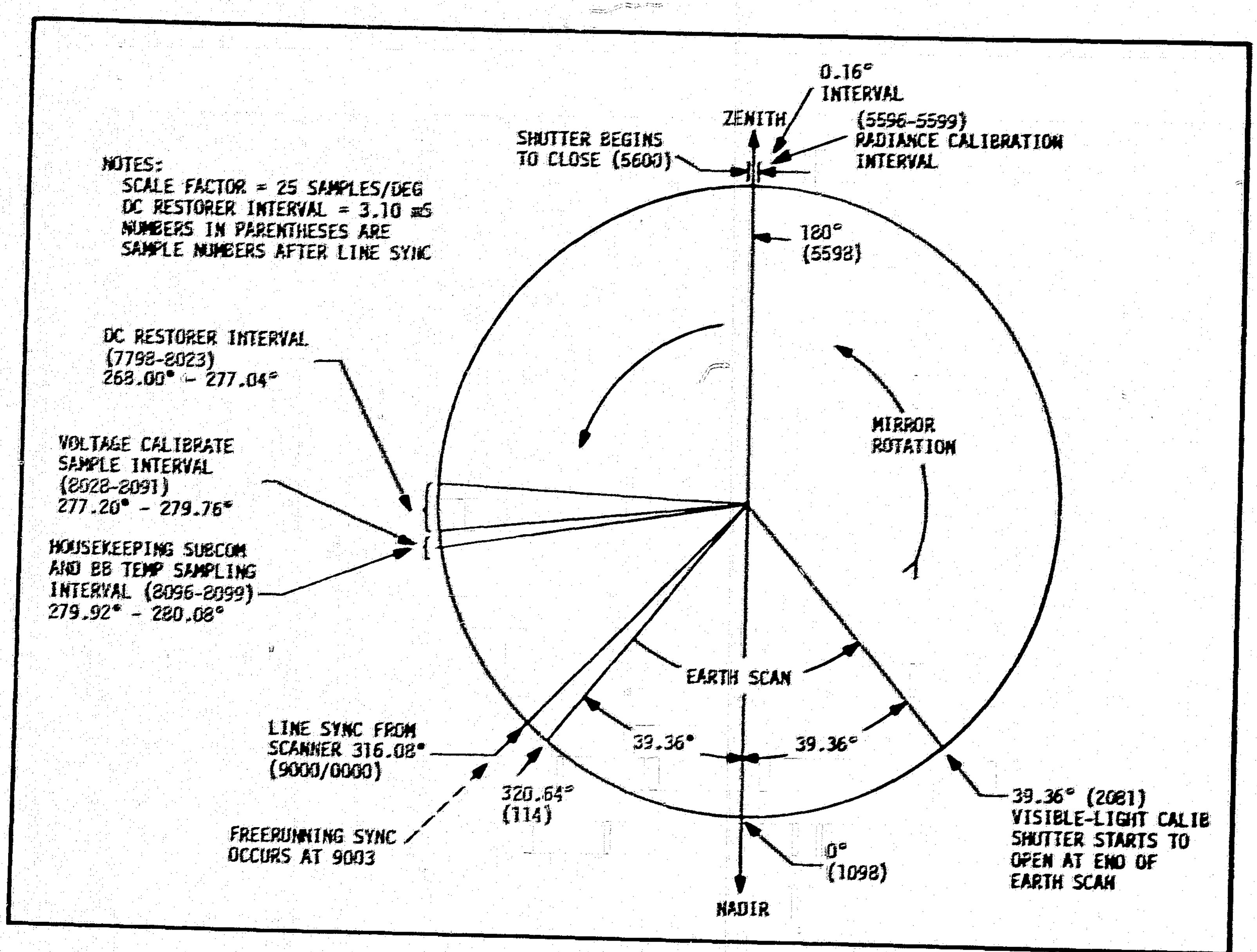


Figure 3.5-4. CZCS Mirror Scan Positions

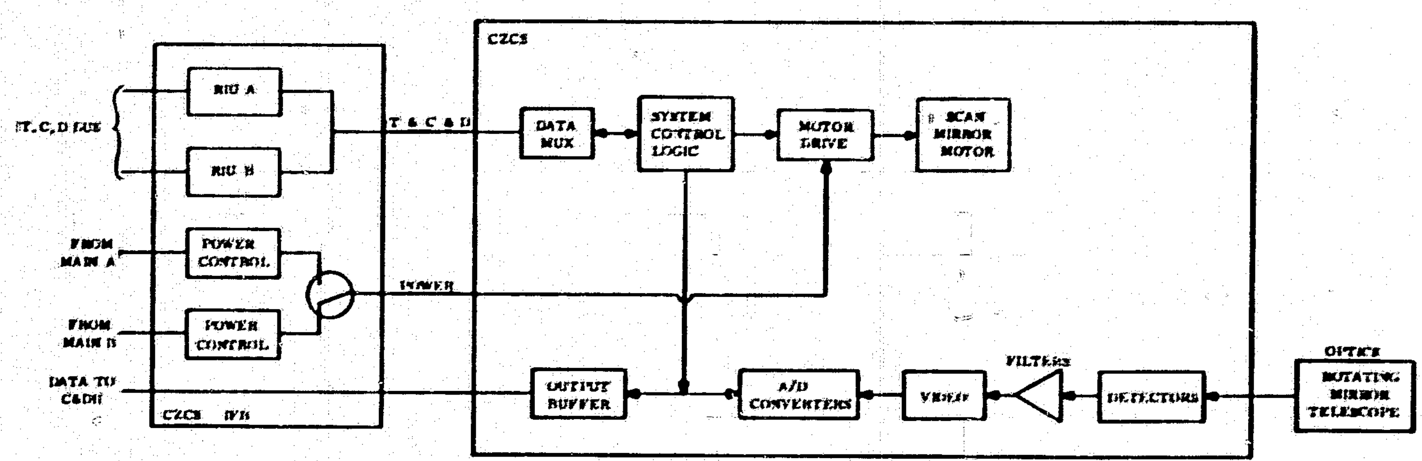


Figure 3.5-5. CZCS BLOCK DIAGRAM

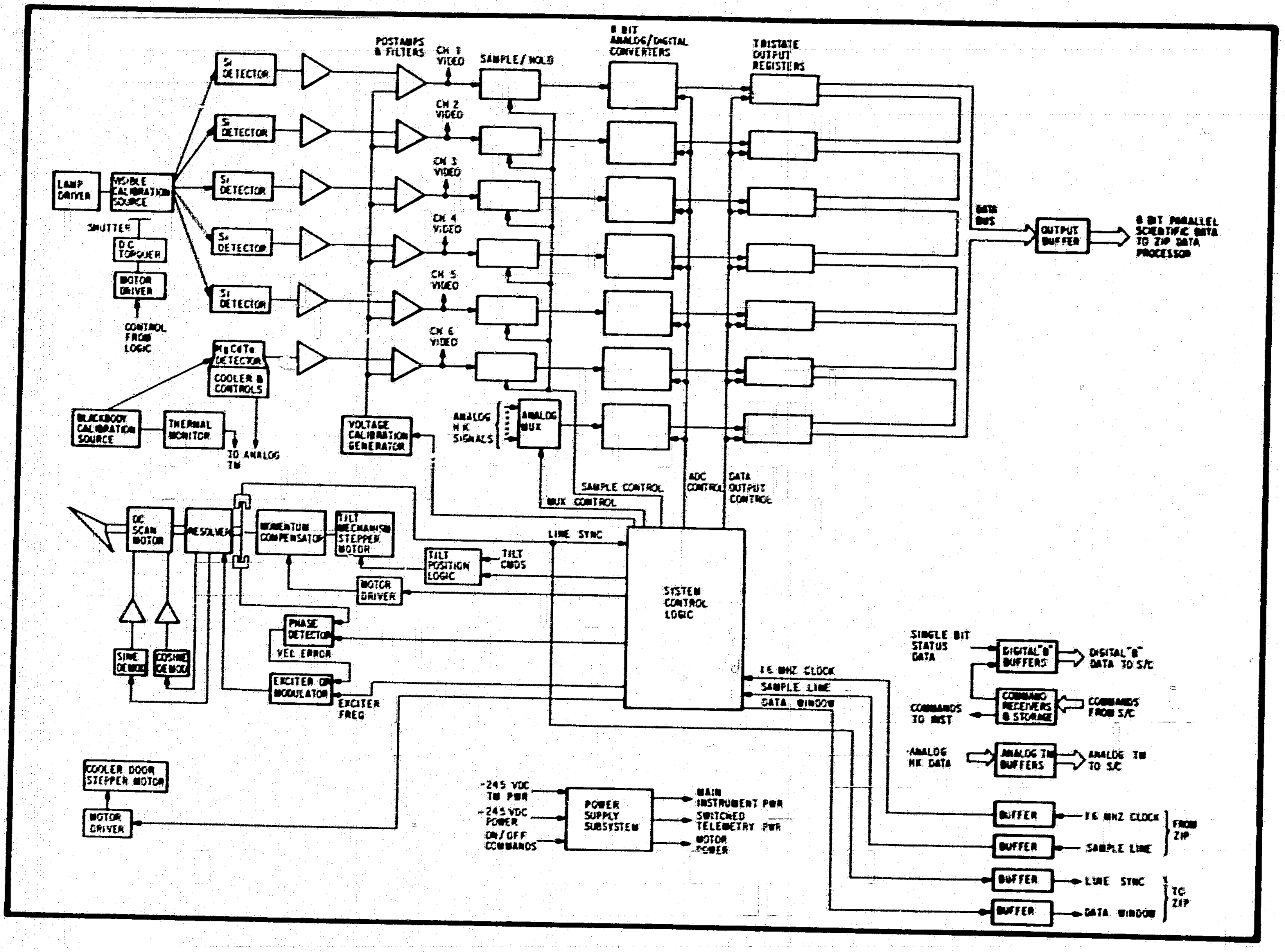


Figure 3.5-6. CZCS System Block Diagram for Nimbus
(the NOSS instrument has nine channels)

TABLE 3.5-1

SIGNIFICANT CZCS CHARACTERISTICS

Weight:	50 Kgm
Dimensions:	85 x 40 x 60 cm
Power:	50 W (Avg), 85 W (Peak) @ -24.5 ± 0.5 V
Scan Rate:	8.08 rps (485 rpm)
Data Scan:	$\pm 39.2^{\circ}$ Cross Track
Scan Tilt:	$\pm 20^{\circ}$ in 2° Increments (along track)
Scan Jitter:	1.8 μs (in vacuum)
Instantaneous Field-Of-View	0.865 mrad (visible) 0.927 x 0.88 mrad (IR)
Cooler Field-Of-View	101° Solid Angle
Maximum Data Rate:	5.24×10^{6} bps
Buffered Data Rate:	1.20×10^{6} bps
Sampling Rate:	72.72×10^{3} samples/second
Sampling Resolution:	8 bits (per channel)
Diameter of Optical Aperature:	17.8 cm
Telemetry:	30 Binary, -0.5 ± 0.5V, -7.5 ± 2.5V: 50 Analog, 0 to -6.375 V
Command:	40 Latching Relays (100-500 Ohm)

3.5.3 INTERFACES

All CZCS spacecraft interfaces, except for wiring to the scanner subsystem launch caging mechanism, are located in the CZCS main electronics subsystem. The intrument requires -24.5 Vdc electrical power, but a change to a +28 Vdc system is being evaluated. Temperature range is 0 to 40°C operating and non-operating.

The instrument utilizes a 1.6 MHz clock and sample line with a pulse period of 13.75 μs provided by a spacecraft information processor. The instrument provides data at a 5.2-Mbps maximum data rate to the information processor which provides a buffering and formatting function for interfacing with the spacecraft tape recorders and downlink. The information processor provides sync words, frame identification words, and time code, and organizes them along with the CZCS data.

3.6 GLOBAL POSITIONING SYSTEM (GPS)

The Global Positioning System (GPS) consists of Navigation Data Satellites (NDS) in nearly-circular, 12-hour orbits and in three orbital planes. Each NDS continuously transmits its position and time (which is periodically updated). The GPS provides precise position, velocity and timing data to NOSS through an on-board GPS Receiver Processor Assembly (RPA).

3.6.1 OBJECTIVE

The NOSS on-board GPS equipment provides position, velocity and timing data which is inserted into the telemetry data stream to aid in positioning each image and into the OBC to aid in vehicle control.

3.6.2 INSTRUMENT DESCRIPTION

A Global Positioning System Receiver Processor Assembly (GPS RPA) provides high-accuracy spacecraft position, velocity and timing data by using signals from four GPS satellites. GPS signals at 1575.42 and 1227.60 MHz with different propagation properties permit determination of ionospheric delay and other medium effects. The on-board system essentially solves four equations in four unknowns (three time-difference-of-arrivals for range and one time correction factor) to compute three position components and time; velocity is extracted by measuring Doppler frequency shift from each of the GPS satellites.

The baseline design uses a single antenna on the space side of the spacecraft. The other GPS RPA components are mounted inside the spacecraft support structure. A simplified block diagram is shown in Figure 3.6-1.

Operation begins with the RPA accepting uplink almanac data via the C&DH subsystem to the RPA memory. The almanac data includes uploading/aiding information containing time, the NOSS orbital parameters, and almanacs of the GPS Navigational Data Satellites (NDS). Discrete commands and serial mode selection commands then initiate RPA action.

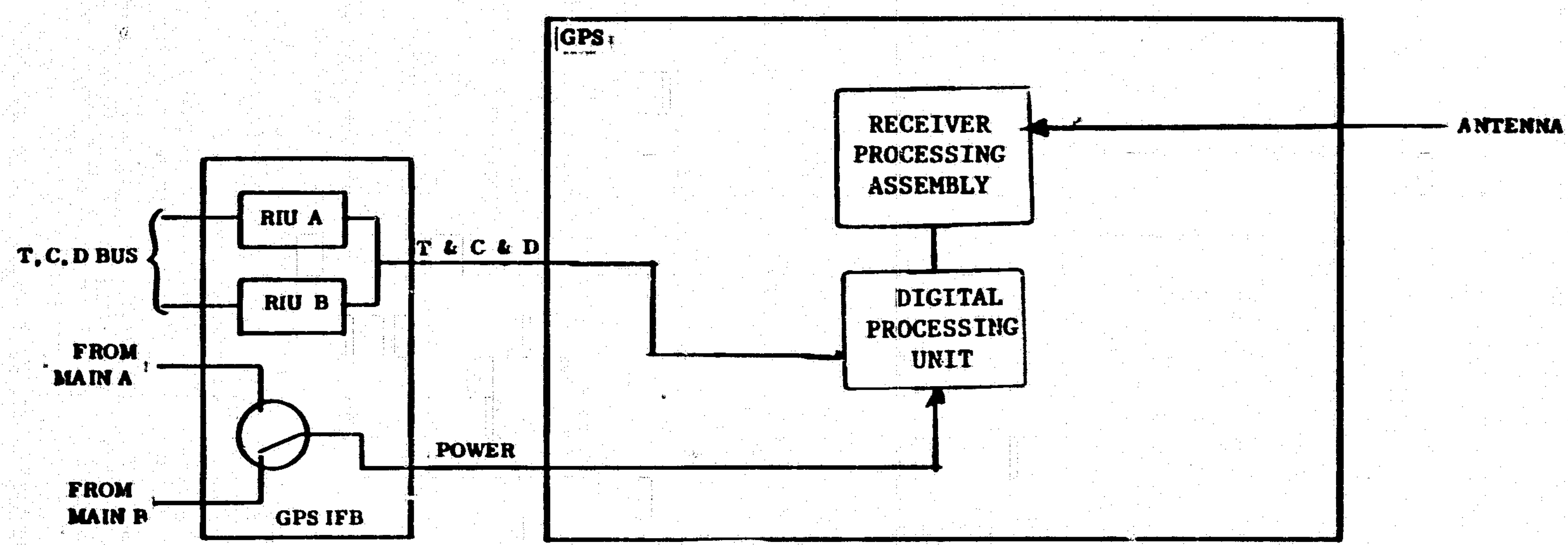


Figure 3.6-1. GPS BLOCK DIAGRAM

The spacecraft then detects and acquires the best geometrically situated rf navigation signals generated by the NDS transmitters. These rf signals are received by the GPS antenna and amplified by the low noise preamplifier/filter assembly. The receiver accepts and operates on the output of the antenna/preamplifier assembly. The receiver includes two identical tracking channels which can be used interchangeably either one at a time, or simultaneously making independent measurements. The receiver outputs the required time, pseudo-range, delta-pseudo-range pairs (L_1 at 1575.42 and L_2 at 1227.60 MHz), and system data extracted from navigation signals to the processor to sequence its operations and aid in the selection and acquisition of the available navigation signals. The processor determines which NDS are within view and if there are more than four, selects the best set of four NDS's for navigation. The RPA is capable, however, of operations with one NDS almanac. The signals to be used are tracked and data extracted.

Because of different relative velocities between NOSS and the various GPS satellites, the two separate quadraphase signals from each NDS satellite undergo different Doppler shifts. Further, each satellite transmits a unique segment of long code and recycles its code segment every 7 days. The phase of the received code, when compared to a reference code, provides data on the range between each GPS satellite and NOSS. This phase difference is determined in the RPA where received and amplified signals from the preamplifier, together with oscillator inputs, are tracked in frequency and in code position. Code shift, thus obtained from four GPA satellites, plus an ephemeris message from each, provides the needed data for the RPA processor to determine the NOSS position, velocity and GPS time. These data are outputted via the Digital Processing Unit (DPU) to all spacecraft user subsystems.

In the absence of NDS signals, the last measured spacecraft position and velocity are extrapolated.

Processing of GPS data which is common to all instruments is accomplished in a GPS-dedicated DPU. These data are then transferred by command from the C&DH module into the C&DH on-board computer concurrent with telemetry readout. GPS position data acquired by the on-board computer is distributed in an abbreviated form to the science instrument RIUs for use in performing footprint computations at each instrument. Other information required for controlling the

instrument and alleviating the ground processing requirements is also computed as appropriate for each sensor.

3.6.3 INTERFACES

Electrical power is provided to the GPS on a +28V (nominal) bus. The GPS requires an L-band antenna, preamplifier, and redundant reference oscillators. The 5.115 MHz oscillator and 1.2 to 1.6 GHz preamp accept power from the receiver/processor assembly.

The GPS can select timing data from either a mission-unique clock subsystem or from the NDS satellites and insert this data in both the telemetry and sensor data streams. Spacecraft position data derived from the GPS is inserted into housekeeping telemetry. Both new GPS data and computed ephemeris data and time are transmitted through telemetry interfaces provided by the spacecraft.

4.0 SUBSYSTEM DESCRIPTION

4.1 POWER SUBSYSTEM

4.1.1 GENERAL DESCRIPTION

The power subsystem configuration selected for the NOSS Mission is based upon the use of two Multimission Modular Spacecraft (MMS) power modules (MPS). Each MPS module is configured with three 50 ampere-hour batteries. To simplify the operation of the power subsystem, each power module works independently of the other deriving its power from a separate solar array section. Load control and switching selection is provided by the Power Switching Distribution and Control Unit (PSDCU). Maximum utilization is made of existing hardware designs - the MPS modules are used without modification and the PSDCU's design is based upon flight-proven hardware. A block diagram of the NOSS power subsystem is shown in Figure 4.1-1 and the subsystem equipment list is presented in Table 4.1-1.

This power subsystem design, while nominally selected to accommodate a three year mission, will support a five year mission with only minimal reductions in performance margins.

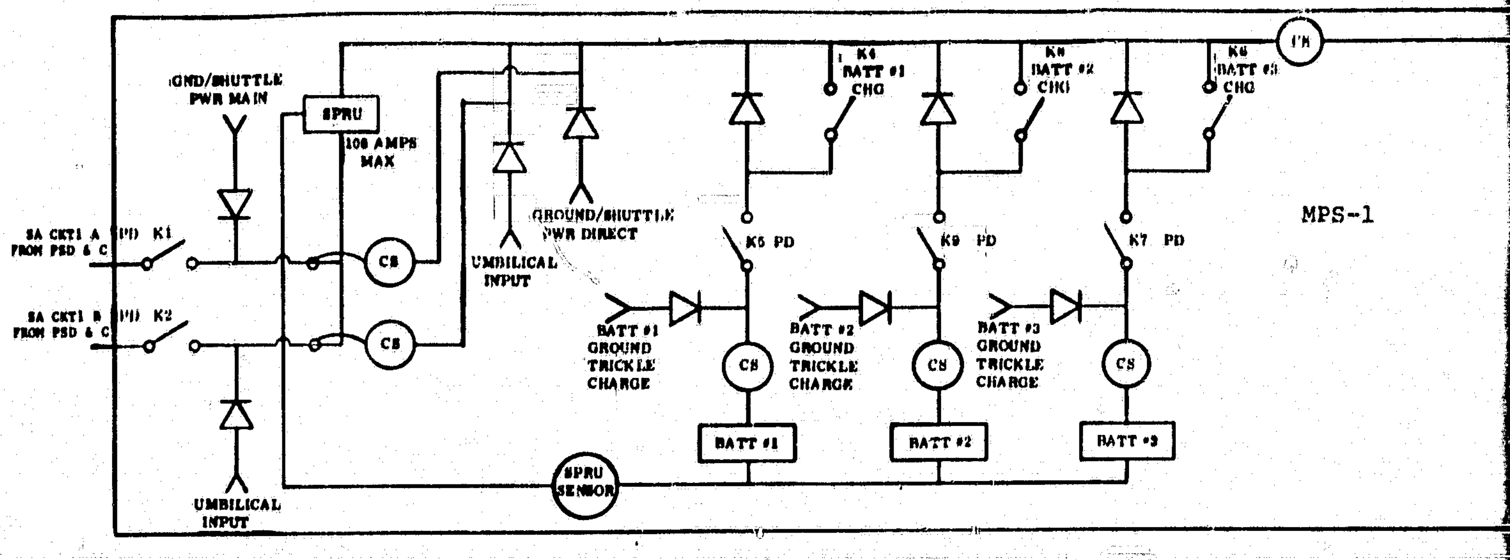
4.1.2 POWER SUBSYSTEM SIZING

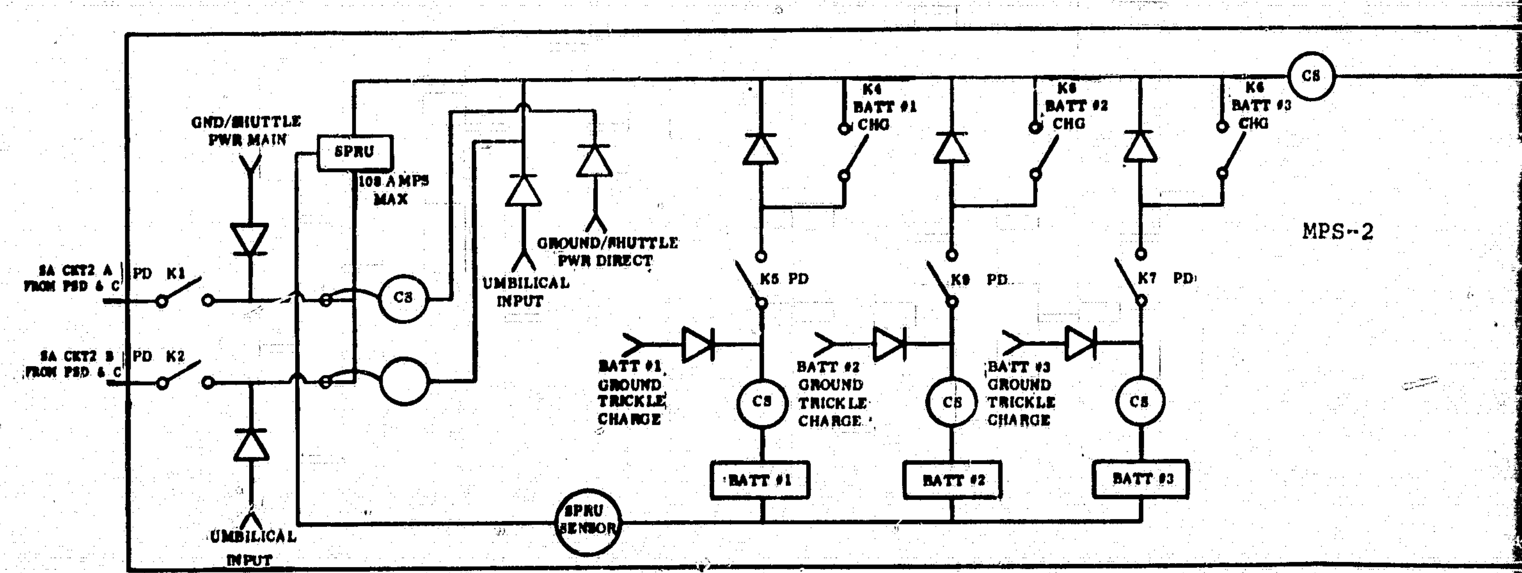
Table 4.1-2 summarizes the assumptions utilized in the determination of the required solar array area, battery complement and power distribution and control power support capabilities. The total load power requirement of 2209 watts, exclusive of battery charging demands, includes a 25% load power growth capability for the instruments and was derived as shown in Table 4.1-3. This load requirement reflects into a total beginning-of-life solar array capability of approximately 7400 watts, at 25°C operating temperature. Similarly, the load demand results in a 16% depth-of-discharge on the six 50 ampere-hour batteries in the two MPS modules. Available flight and ground test data suggests useful battery lifetimes in excess of ten years at battery operating temperatures of 0-5°C, and at least five years at temperatures in the 15°C region.

TABLE 4.1-1

POWER SUBSYSTEM EQUIPMENT LIST

	Weight	Power	Size	Telemetry		Command
	KG	Watts	L x W x H (in)	Analog	Digital	Pulse
Solar Array	107.7	7,400 (output)	Ret 8" x 152" x 40" Ext 46.5' x 152" Cylinder	20	8	16
Solar Array Drive	10.4	40	17.6" x 8.15"	6	8	16
Electronics Pk #1	4.5		8" x 7" x 4.25"			
Electronics Pk #2			8" x 7" x 3.5"			
Power Dist. SW and Contr. Unit	45.4	30	24" x 24" x 12"	34	36	72
MPS #1	272.2	50	47" x 47" x 18"	41	8	46
MPS #2	272.2	50	47" x 47" x 18"	41	8	46
Subtotal	712.4	170		142	68	196
LAMMR I/F	10.9	54	12" x 8" x 8"	-	32	64
CZCS I/F	5.4	13			16	32
Alt I/F	10.9	20	12" x 8" x 8"	-	16	32
SCAT I/F	10.9	29	12" x 8" x 8"	-	36	72
Subtotal	38.1	116		-	100	200
Totals	750.5	402		142	168	396





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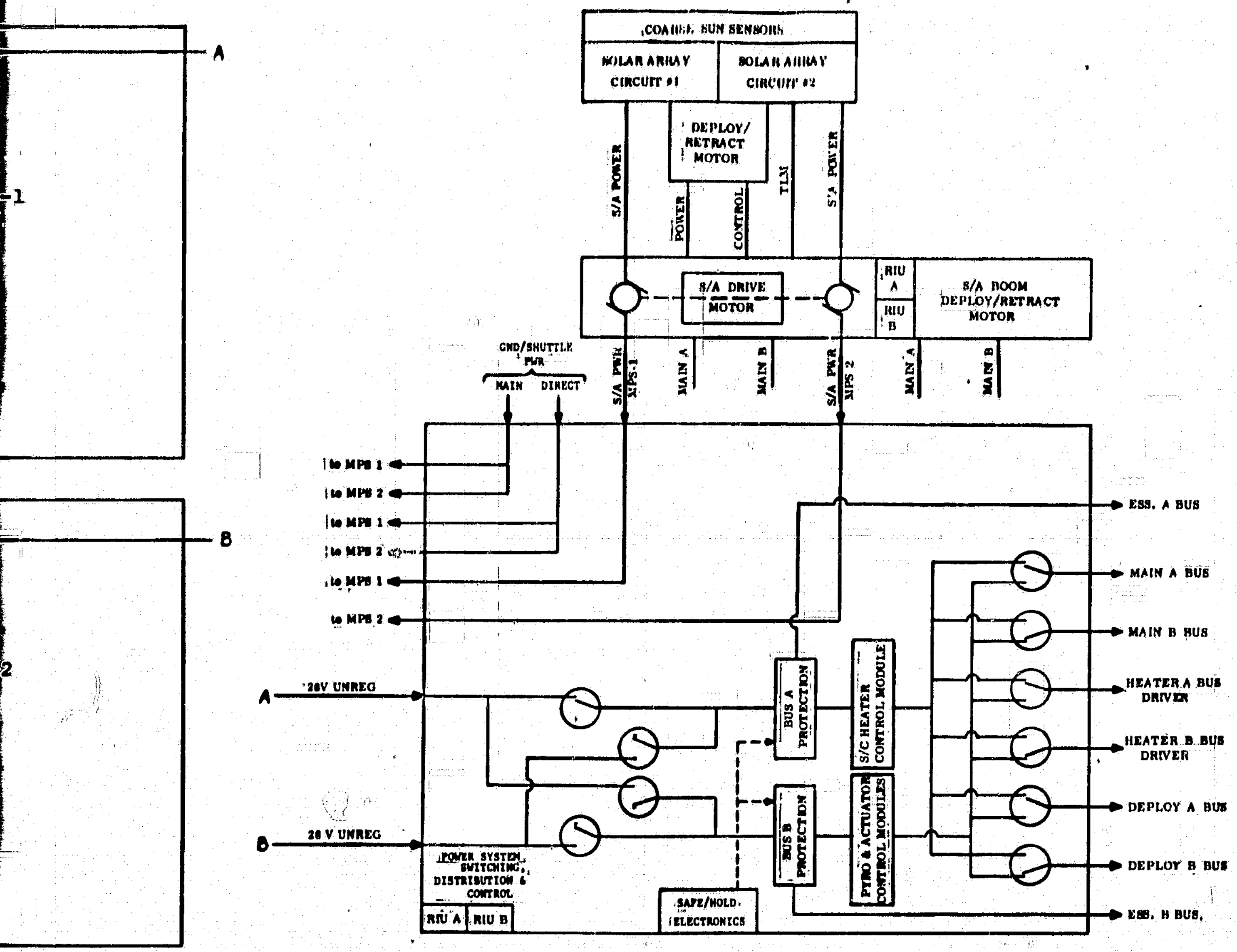

Figure 4.1-1. Power Subsystem Block Diagram

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Table 4.1-2. POWER SYSTEM DESIGN REQUIREMENTS

A. Load Power Requirements - 2209 Watts

- o 25% instrument growth
- o Regulation/inversion equipments provided by spacecraft
- o Worst-case spacecraft load demands

B. Power Interface Requirements

Instrument	Interface	Power Level (watts)
LAMMR	+28 ± .5 VDC	350
CZCS	-24.5 ± .5 VDC	50 (Avg.) 85 (Peak)
ALT	+24 to +34.5 VDC	177
SCAT	+28 ± .3 VDC +28 ± 4.0 VDC	90 (Avg. 75 (Avg) 310 (Peak)
GPS	21-35 VDC	35

C. Solar Array Requirements

1) EOL: PSA = 2,209 x 1.89*= 4,175 W (EOL, 75°C)

*1.89 = Conversion efficiency factor for the MPS for the NOSS orbit. This factor is derived from an energy balance for the solar array supplying power for the daytime load as well as recharging the batteries following the eclipse period.

2) BOL: PSA = 5,567 W ÷ 0.77* = 7,230 W (BOL, 25°C)

* Array degradation factors assumed

- 2% Thermal Cycling
- 2% Ultraviolet
- 14% Charged Particle Irradiation
- 5% Uncertainty due to orbital and temperature variations

3) Solar Array: 54.7 M^2 (135 w/m^2)

D. Battery Capacity Requirements - for five years reliable battery performance lifetime in 10 - 20°C operating region, maintain battery DOD in 20% range.

1) Required Batt. Capacity

$$= \frac{P_{load} \times T_n}{V_{Batt} \times X\ DOD}$$

$$= \frac{2209 \times 35\%\ min}{26.4\ v \times .2}$$

$$= 244.05\ AH$$

2) Battery Capacity = 6 x 50 AH Batteries (300 AH)

$\longrightarrow$ 16.0% DOD

E. Power Distribution Capacity

1) Required: 2,209 W

2) 2 MPS Module Capacity: 2,400 W

TABLE 4.1-3

POWER SUMMARY

SUBSYSTEM	POWER - WATTS
Thermal	400
Power	
MPS	100
PSDCU	30
Solar Array Drive	40
ACS	
MACS	95
Star Tracker	46
ESA	15
Propulsion	135
C&DH	
C&DH	100
High Rate Data Storage	48
HGAS	40
GPS	35
Encryption	20
Instruments	
LAMMR	404 (350*)
SCAT	194 (165*)
ALT	197 (177*)
CZCS	63 (50*)
LRA	
25% Instrument Growth	215
Contingency	32
Total	2,209 W

* LAMMR 285W Reg ÷ .85 = 335.3
65W Unreg. (but max 34 volts) ÷ .95 = 68.5 〉 404 Total Load Bus Power

* SCAT 165W Reg ÷ .85 = 194 W 〉 Total Load Bus Power

* ALT 177W Semi Reg. ÷ .9 = 197W 〉 Total Load Bus Power

* CZCS 50W Reg ÷ .80 = 63W 〉 Total Load Bus Power

4.1.3 MODULAR POWER SYSTEM (MPS) DESCRIPTION

The MPS provides the major functions for energy storage and power control of unregulated +28 VDC to the spacecraft bus. Power from a mission unique solar array is supplied to the MPS via the Module/Structure Interface Connector. A MPS block diagram is included in Figure 4.1-1.

Provisions are included for receiving power from external power sources during any mission phase from pre-launch (ground) operations through on-orbit resupply or retrieval operations. Isolation diodes, contained in the MPS, are used to prevent faults on the external power lines from damaging MPS equipment. Any conditioning of the unregulated +28 VDC power to other power types, voltage levels, or regulation tolerances is provided by unique power conditioning interface adapter units associated with each instrument.

The MPS accepts unregulated DC power from the solar array. The MPS contains power electronics to control this solar array energy, to supply the load bus power requirements, and to control battery charge currents and voltages. Nickle-cadmium batteries are used to supplement solar array power during periods when solar array power is unavailable or insufficient to support the spacecraft load. Battery charging is accomplished whenever solar array power capability exceeds the spacecraft load demand. The NOSS MPS design uses three 50 AH batteries for each MPS.

After experiencing a discharge period, the batteries automatically enter a charge mode, provided sufficient solar array power is available. The primary battery charge mode includes an initial charge phase followed by a voltage limited, taper charge phase. During initial charging, the batteries receive all available solar array energy in excess of the load demand. This phase of charging is terminated when the battery voltages rise to a preselected battery voltage limit. Eight temperature compensated voltage limits are provided for battery charge control with the operating level selectable by command. Upon reaching the selected voltage limit, the battery charge currents are reduced (tapered) to maintain the battery voltage at the temperature compensated level.

Battery charging/discharging is accomplished in the parallel mode for all cases. This requires that temperature differences between batteries be held to 10°C or less and that impedance mismatch of the charge/discharge paths be minimized.

4.1.4 POWER SWITCHING, DISTRIBUTION AND CONTROL UNIT (PSDCU)

The PSDCU shown in Figure 4.1-1, is the primary interface unit for all of the power subsystem functions. All MPS input power, either from the solar array or from ground/shuttle input sources, passes through the PSDCU. The PSDCU also provides two isolated power distribution busses, each with its own bus protection circuitry, for load power support. The PSDCU also contains the safe-hold electronics, which removes power to all non-essentail NOSS subsystem instruments in the event of on-board anomalous behaviour that can result in danger to the spacecraft.

Modules from the MMS signal conditioning and control unit (SC&CU) are provided in the PSDCU to perform the following two basic functions:

(a) Standard driver circuits and armable power buses for mission-unique pyro and actuator circuits

(b) Control of heaters mounted on the structure and monitoring of the structure and solar array temperatures.

4.1.5 LOAD CONTROL AND REGULATION

Each single instrument is provided with a redundant power interface unit, located as close as possible to the individual instrument package. This unit provides a means by which each instrument can be connected to or removed from the MPS power source, protects the other power users from faults within a given instrument and supplies conditional power for those users who are unable to use the MPS 28V $\pm$ 7 VDC power directly. The size and mass of each instrument power interface unit is specified in Table 4.1-1.

4.1.6 DEPLOYABLE/RETRACTABLE SOLAR ARRAY

A single, modified Power Extension Package (PEP) solar array has been chosen for the NOSS mission. The PEP is a integral dual winged solar array designed for shuttle environments and is to be flight qualified by 1983. The basic design data for the PEP array are as follows:

Si Cells	-	8 mils thick
	-	2 x 4 cm wrap around
	-	12.8% efficiency
Cover	-	6 mil microsheet
Substrate	-	1/2 mil kapton with internal wiring and welded interconnects
Wing size/ area	-	3.84 x 37.8 meters/145 meters^2
Output of wing	-	16.4 KW
Output of PEP	-	32.8 KW

There are three modifications required to the PEP, all down scaling or simplifications, to accommodate the NOSS mission. These modifications are as follows:

1. The canister containing the extendable boom is mounted parallel to the long axis of the solar array storage box and rotated 90° for solar array deployment in the PEP configuration. In the NOSS configuration the extendable boom canister is hard mounted in the solar array deployment position eliminating the need to rotate the canister 90°.

2. The PEP solar array utilizes two of the units shown in Figure 4.1-2 mounted back to back at the non-deployable side of the solar array storage box (dual wings). The PEP solar array deployment takes place in opposite directions (180^{o} apart). The NOSS solar array is a single extension unit (single winged) as shown in Figure 4.1-2. This modification may be described as 1/2 of a PEP solar array.

3. The length of the remaining one wing of the extended solar array is shortened from 115 feet in the PEP configuration to approximately 46.5 feet for the NOSS configuration.

The beginning of life solar array power requirement has been calculated to be 7.4KW, degrading to 4.2 KW at 75^{o}C at the end of a three year mission. Total solar array area required is 54.7 m^{2}. Total mass of the modified PEP unit is 107.7 Kg which includes substrate, deployment/retraction motors, mechanisms, and sensors.

4.1.7 SOLAR ARRAY DRIVE

The Ball Brothers Model EMS 251 Solar Array Drive Assembly (Figure 4.1-3) has been chosen for the study. The EMS-251 is an existing design incorporating slip rings for power and control signal transfer. Multiple slip rings exist for the transfer of power to assure the separation of multiple power buses and returns. The unit is redundant except for the shaft and the bearings. The design is available in a brush or brushless motor configuration with an open or closed loop control system. A brushless type motor and a closed control system with a sun tracking accuracy of ± 1 degree have been selected for the NOSS mission. The EMS 251 is configured as a cylinder with a length of 44.7 cm and a diameter of 20.7 cm. Its mass is 10.4 Kg. The associated electronics packages are configured in two units with dimensions of 20.3 cm x 18.4 cm x 12.1 cm and 20.3 cm x 18.4 cm x 8.9 cm with a total mass of 4.54 Kg.

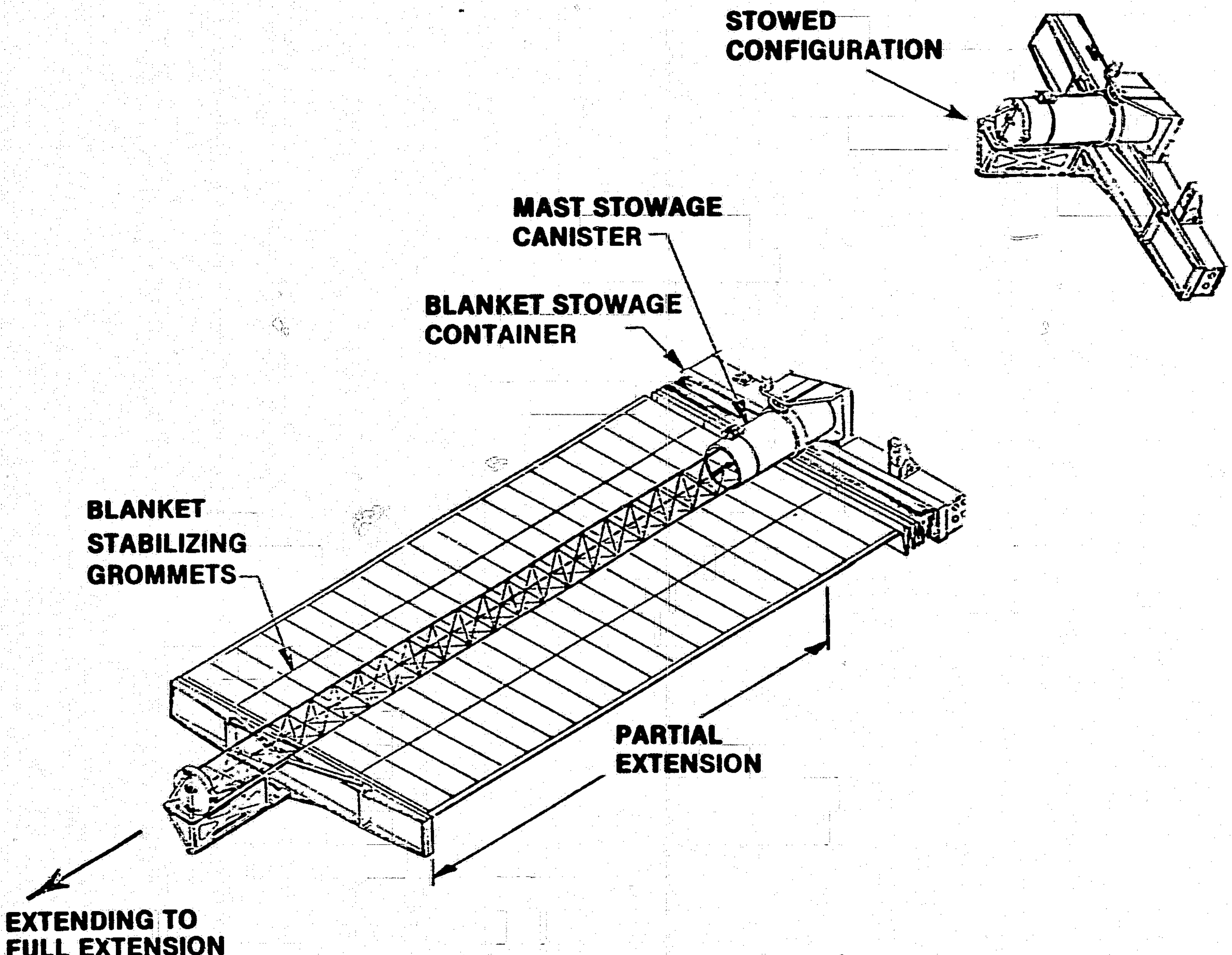


Figure 4.1-2 Modified PEP Solar Array

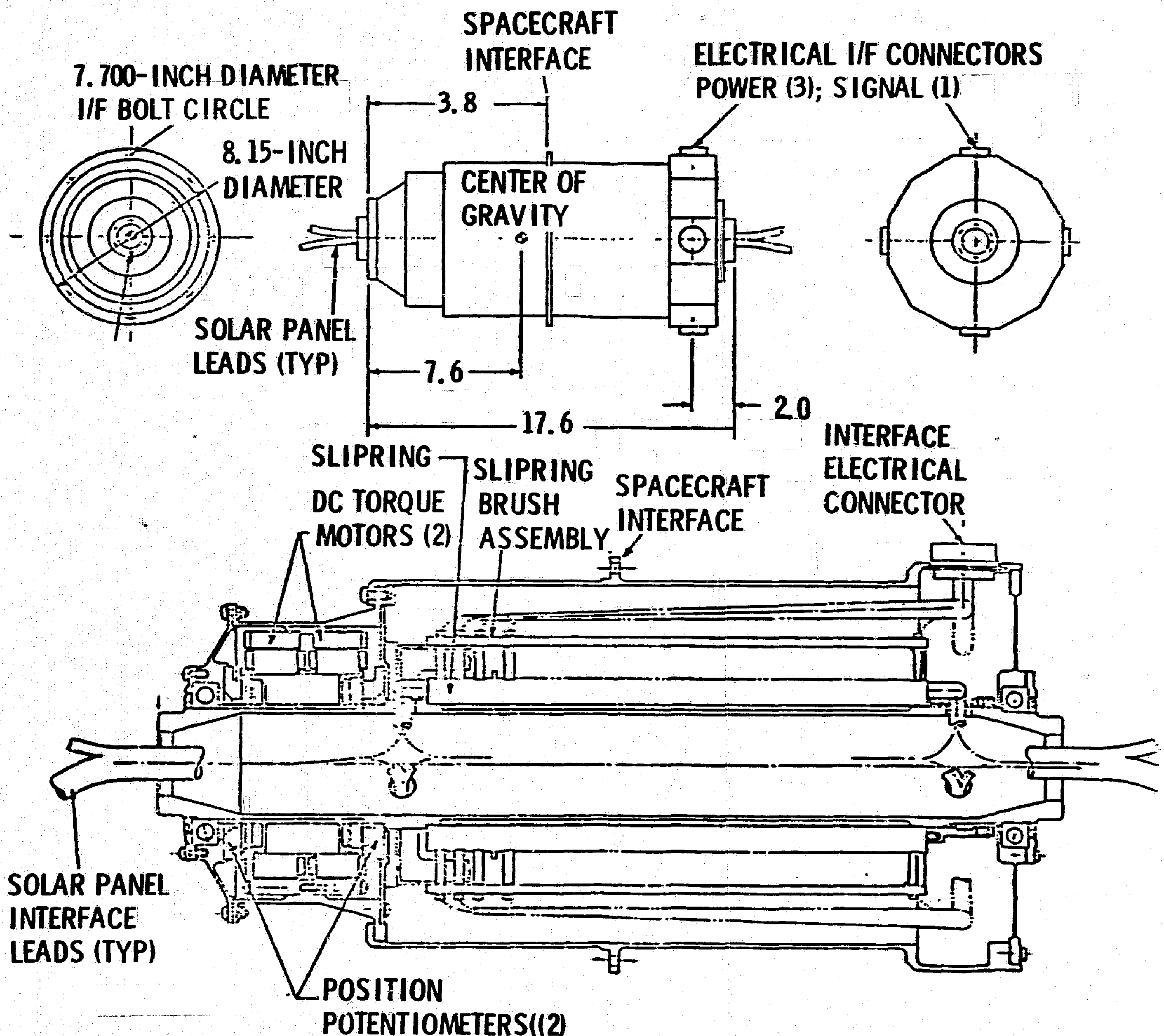


Figure 4.1-3. Solar Array Drive

4.2 NOSS PROPULSION SUBSYSTEM (NPS)

4.2.1 GENERAL DESCRIPTION

The NOSS Propulsion Subsystem (NPS) contains a single central 1.9-meter (76-inch) diameter screen propellant tank which feeds four thruster complements. Each complement contains one 200 N (45 lb) orbit transfer and four 2.2N (.5 lb) attitude control hydrazine thrusters. Modules are pressure fed over a 3:1 blowdown ratio through separate filters and latch valves from the central tank as shown in the system schematic (Figure 4.2-1). The latch valves combine with series redundant thruster valves to provide a triple seal along the flow path to the thrusters as dictated by shuttle safety requirements. Fill and drain valves for loading, pressure and temperature transducers for status monitoring, line and tank heaters for thermal control, and electrical harness and miscellaneous lines, fittings and structure complete the system. The subsystem equipment list is summarized in Table 4.2-1.

A nominal propellant load of 1724 Kg provides sufficient impulse to: transfer from the 300 km shuttle parking orbit to the 800 km circular NOSS orbit and return, provide 36.4 m/sec ΔV for on-orbit maneuvering, and provide 4.4 N-sec impulse per hour for momentum wheel unloading, rate nulling and attitude control. The system can accommodate up to a 2^{o} plane change maneuver if loaded to its maximum capacity (2509 Kg hydrazine @ 3:1 blowdown).

The orbit transfer and attitude control thrusters are arranged in modules on the +Y and -Y faces of the spacecraft as shown in Figure 4.2-2. Primary and redundant vehicle control maneuvers are accomplished according to the hierarchy listed on the figure. Orbit transfer attitude control is accomplished by off-pulsing the appropriate thruster to maintain the vehicle within attitude and attitude rate limits. Attitude control thrusters are commanded in force couples to avoid large cross-coupling torques.

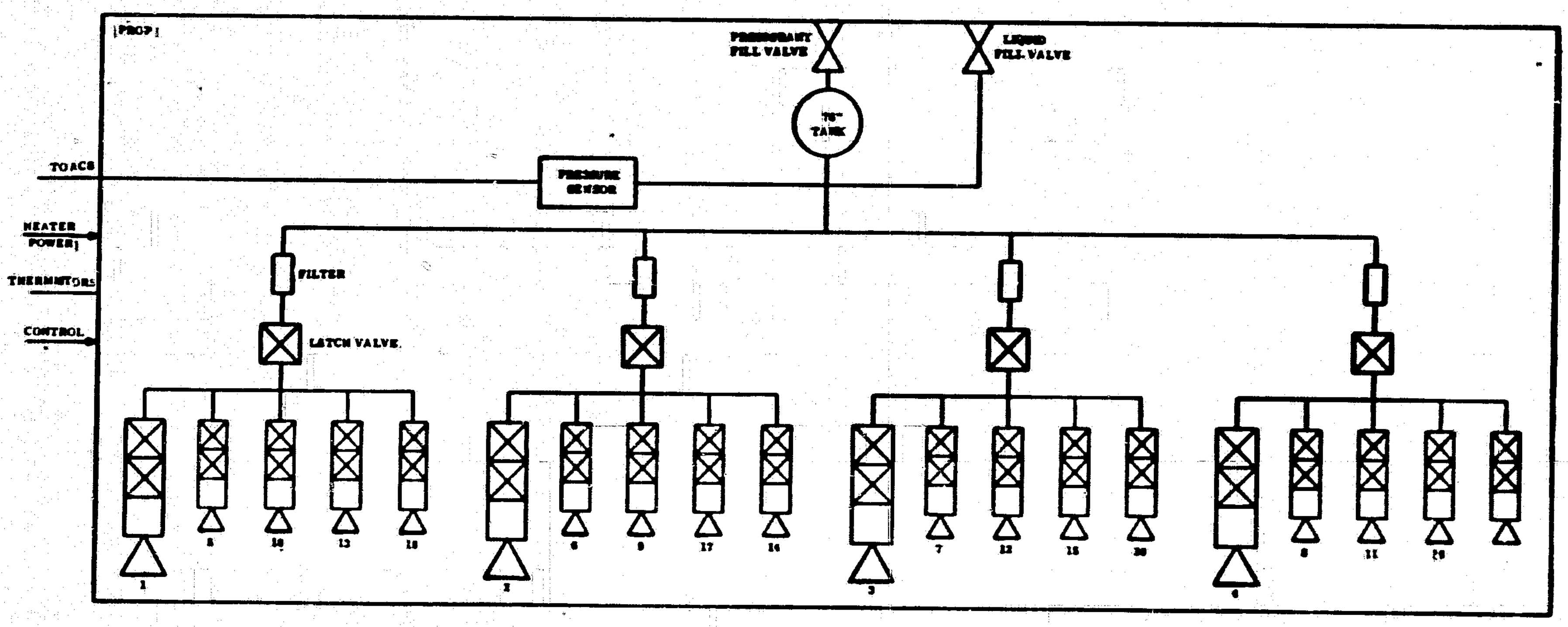
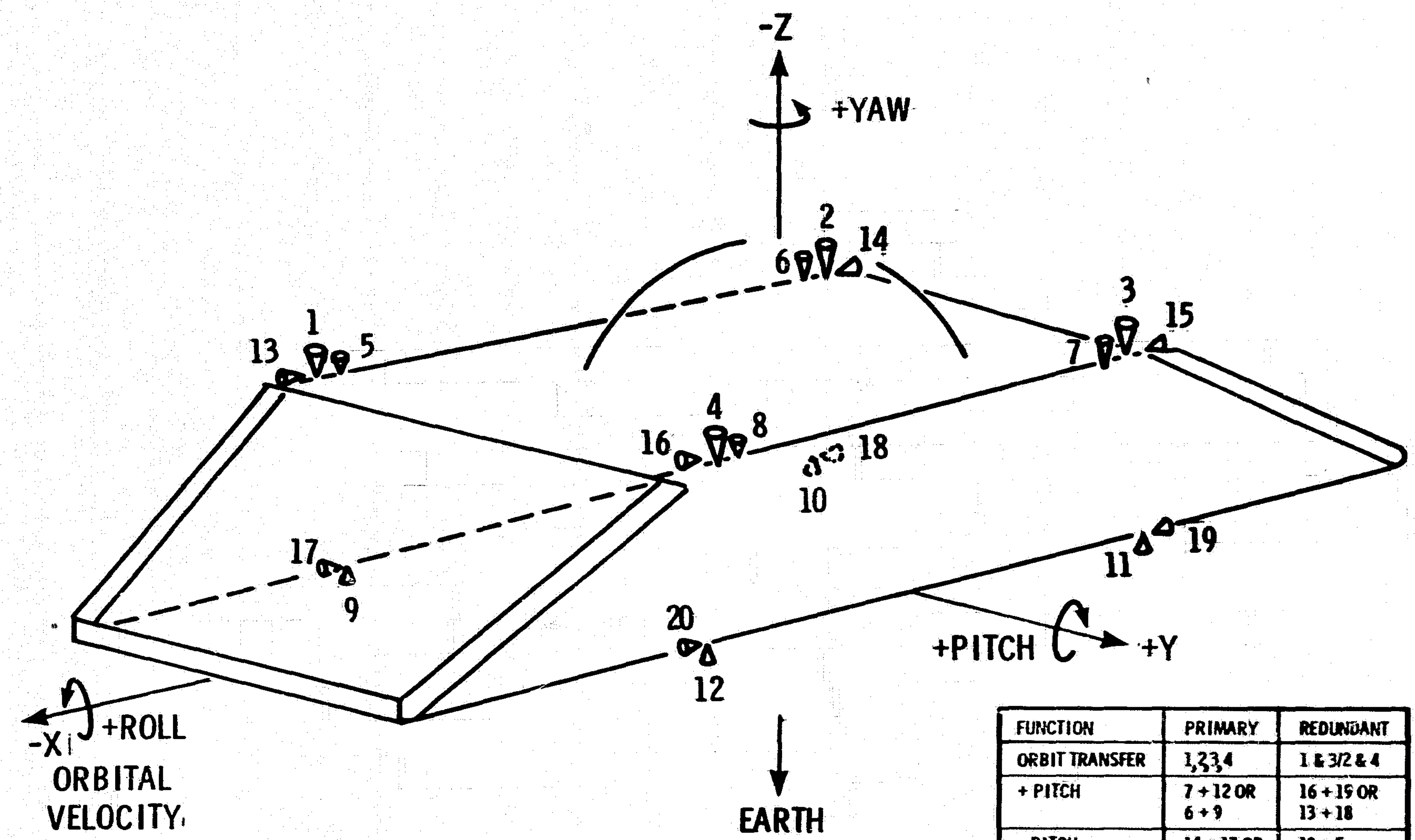


Figure 4.2-1 NPS Schematic Diagram

TABLE 4.2-1

NOSS PROPULSION SUBSYSTEM EQUIPMENT LIST

Item	Quantity Number	Mass Kg	Power Watts	Size L x W x H	Telemetry		Command		
					Analog	Digital	Pulse	Level	Serial
Tank	1	211.0		76" Dia. Tank					
Thruster Assembly					20	25	10	20	20
o 45 lb.	4	10.9							
o 0.5 lb.	16	5.1	(Control in ACS Module)						
Latch Valve	4	2.4				4	8		
Fill/Drain/Vent	2	.3							
Filter	4	1.5							
Sensors									
o Pressure	1	.2			1				
o Temperature	–				20				
Blanket/Heaters	–	12.7	135			20	34		
Harness and Connector	–	10.9							
Junction Box	12	5.4							
Thruster Enclosure Box	8	1.8							
Tubing and Fittings	–	6.8							
TOTAL DRY	–	269	135						
Propellant		1,724							
TOTAL WET		1,993	135		41	49	52	20	



ORBIT TRANSFER = 45 LB_f

ATT. CONTROL = 0.5 LB_f

FUNCTION	PRIMARY	REDUNDANT
ORBIT TRANSFER	1,2,3,4	1 & 3/2 & 4
+ PITCH	7 + 12 OR 6 + 9	16 + 19 OR 13 + 18
- PITCH	14 + 17 OR 15 + 20	10 + 5 11 + 8
+ ROLL	5 + 12	6 + 11
- ROLL	9 + 8	10 + 7
+ YAW	16 + 14	20 + 18
- YAW	13 + 15	17 + 19

Figure 4.2-2 Thruster Arrangement

4.2.2 DETAILED DESCRIPTION

In arriving at the detailed design the establishment of propellant requirements and the resulting selection of a tankage concept were the key design drivers. Thruster sizing, vehicle thermal control and system mechanical integration were the remaining considerations.

4.2.2.1 Propellant Requirements

Propellant is required by the NOSS mission to provide impulse for orbit transfer, on-orbit maneuvers and attitude control.

Orbit transfer from the 300 km shuttle parking oribt to an 800 km circular spacecraft orbit requires perigee and apogee Δ Vs of 139 m/sec and 136 m/sec when accomplished by a Hohmann type transfer. An additional 5 percent is added to the ideal Δ Vs to account for the low thrust to weight and resulting non-impulsive nature of the manuevers (Ref. 1). Return Δ V requirements are identical. Using these Δ V requirements, a dry spacecraft weight of 4838 Kg, a thruster Isp of 2250 Ns/Kg and an iterative impulsive Δ V calculation results in the required oribt transfer propellant quantities shown in Table 4.2-2.

Also shown is the propellant quantity resulting from an allocation of 36.4 m/sec Δ V for the maintenance of a constant period sun synchronous orbit. This results from yearly manuevers to correct inclination ($\sim$.056 ^{o}/year) or trim apogee/perigee.

Attitude control propellant was calculated from an allocation of 4.4 N sec impulse per hour over a 5 year mission and a pulsing thruster Isp of 1450 Ns/Kg. The impulse is based on ten 0.1 sec pulses per hour from a pair of attitude control thrusters and are used for momentum wheel unloading, rate nulling and for a limited amount of jet-only attitude control.

The miscellanepus propellant was included for trapped propellant and expulsion residuals. Additional propellant for flight contingencies was not budgeted.

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[1] AIAA Paper 78-1094,"Analysis and Design Study of the MMS Hydrazine Propulsion Module".

Table 4.2-2 NOSS Propellant Requirements

	Nominal Load		Maximum Load		
	kg	(lbs)	kg	(lbs)	
ΔV_1	416.3	(917.7)	466.9	(1029.3)	
ΔV_2	382.7	(843.8)	429.1	(946.1)	
Attitude Control	132.4	(291.9)	132.4	(291.9)	
On Orbit ΔV	90.8	(200.2)	102.1	(225.0)	
Plane Change	----	----	677.4	(1493.4)	[2^{o}]
ΔV_3	344.4	(759.3)	344.4	(759.3)	
ΔV_4	328.5	(724.2)	328.5	(724.2)	
Miscellaneous	28.6	(63.0)	28.6	(63.0)	
TOTAL	1723.7	(3800)	2509.3	(5532)	

The total propellant load of 1724 Kg requires tankage of greater than 2.5 m^3 capacity. An existing 1.93 meter diameter tank with $3.77m^3$ capacity was found which provided a maximum propellant load of 2509 Kg at a 3:1 blowdown ratio. Re-examination of the up-orbit transfer propellant requirements for the increased load left sufficient propellant to acomplish up to a 2^o plane change as can be seen from the summary in Table 4.2-2.

4.2.2.2 Tankage

Table 4.2-3 shows the availabe hydrazine tanks which were considered for NOSS. The 1.9 meter diameter tank is currently being developed by Lockheed for a classified program. It is a scaled-up version of a 1.6 meter diameter tank which has flown successfully on several missions. These tanks utilize capillary propellant management systems.

The TDRS tank uses an AF E 332 diaphragm for controlling propellant orientation. Due to its relatively small capacity, the tank would require a multiple tank installation (at least six) to satisfy the NOSS capacity requirements and was therefore not considered further. The remaining choice between two 1.6 meter or one 1.9 meter tanks resulted in the baseline selection of the single tank based on less complexity, higher reliability, allowance for weight increases and provision for flight contingencies.

4.2.2.3 Thrusters

Figure 4.2-2 shows the arrangement of thrusters on the spacecraft. Four (4) orbit transfer and sixteen (16) attitude control thrusters are arranged in modules on the +Y and -Y faces. The control axes shown correspond to the in-orbit path of the vehicle.

The attitude control thrusters have a nominal thrust of 2.2N and are used to unload momentum wheels, re-orient the spacecraft, and to provide attitude control of the vehicle during orbit transfer maneuvers. Due to the available locations for these units, they are commanded on as force couples to avoid cross-coupling torques into other axes.

Table 4.2-3 Available Hydrazine Tanks

	62"	TDRS (40" W)	76"
Hydrazine Capability, kg (lbs)	2041(4500)	458(1010)	3765(8300)
3:1 Blowdown, kg (lbs)	1362(3003)	305(673)	2509(5532)
Source	Lockheed	TRW	Lockheed
Status	Flight Proven	Flight Qualified	In Development
Prop. Management	Capillary	Diaphragm	Capillary/Baffles

Prior to the orbit transfer manuever, the attitude control thrusters are used to pitch the spacecraft ± 90 degrees to align the spacecraft Z axis with the velocity vector. Four 200N orbit transfer thrusters are then commanded on to obtain the desired ΔV. The resulting 800 N thrust is sufficient to achieve an orbit transfer time of 60 minutes.

During the orbit transfer maneuver, the on-board guidance and control system off-pulses the appropriate orbit transfer thruster(s) to maintain the vehicle within the allowable control switch limits (attitude and attitude rate). For example, using a thruster command pulse width of 0.2 seconds, a maximum vehicle rate of 1.6°/sec. will result if two of the orbit control thrusters are pulsed to correct roll errors. Since the saturation rate for the gyros is 2°/sec. this method of control is feasible for the baselined systems.

Additional vehicle control is provided during the orbit transfer maneuver by using the attitude control thrusters in pairs, as necessary. The attitude thruster couples result in a 3:1 control margin over representative orbit thruster misalignments (two orbit control thrusters misaligned by 0.1° about the orbit transfer roll axis).

4.2.2.4 Mass Properties

Table 4.2-1 lists a weight breakdown of the NPS components. The weight of the 1.9-m diameter tank was obtained by scaling the currently available 1.6-m tank.

4.3 NOSS ATTITUDE CONTROL SUBSYSTEM

4.3.1 GENERAL DESCRIPTION

The NOSS Attitude Control System (ACS) is an adaptation of the Modular Attitude Control System (MACS) used on the Multimission Modular Spacecraft (MMS). A block diagram of the NOSS Attitude Control System is shown in Figure 4.3-1.and the location of the major components on the spacecraft is shown in Figure 4.3-2.

The system consists of an Inertial Reference Unit (IRU), 2 Fixed Head Star Trackers (FHST), a Fine Sun Sensor (FSS), 8 Coarse Sun Sensors (CSS), an Earth Sensor (ES), 4 Reaction Wheels, 20 Hydrazine Thrusters (from the propulsion subsystem), 2 Remote Interface Units (RIU), and Electronic Boxes (analog control and sensor electronics) and usage of the NOSS On-Board Computer (OBC).

4.3.2 SUBSYSTEM DESIGN REQUIREMENTS

The NOSS is an Earth-pointed spacecraft controlled to nadir within ± 0.2 degrees in all axes with the +Z axis aligned to Earth nadir and the -X axis pointed into the velocity vector. On-board attitude determination is computed to ± 0.1 degrees and on-ground attitude determination must be provided to $\pm$ 0.03 degrees. Additional Assumption/Requirements used to generate this preliminary design of the NOSS ACS are:

1. Shuttle launch from WTR to a 300 km, 98.6° inclination circular orbit.

2. Final orbit to be achieved by NOSS propulsion is 800 km, Sun synchronous (98.6° inclination), circular with the Sun at 10:30 a.m., ascending node.

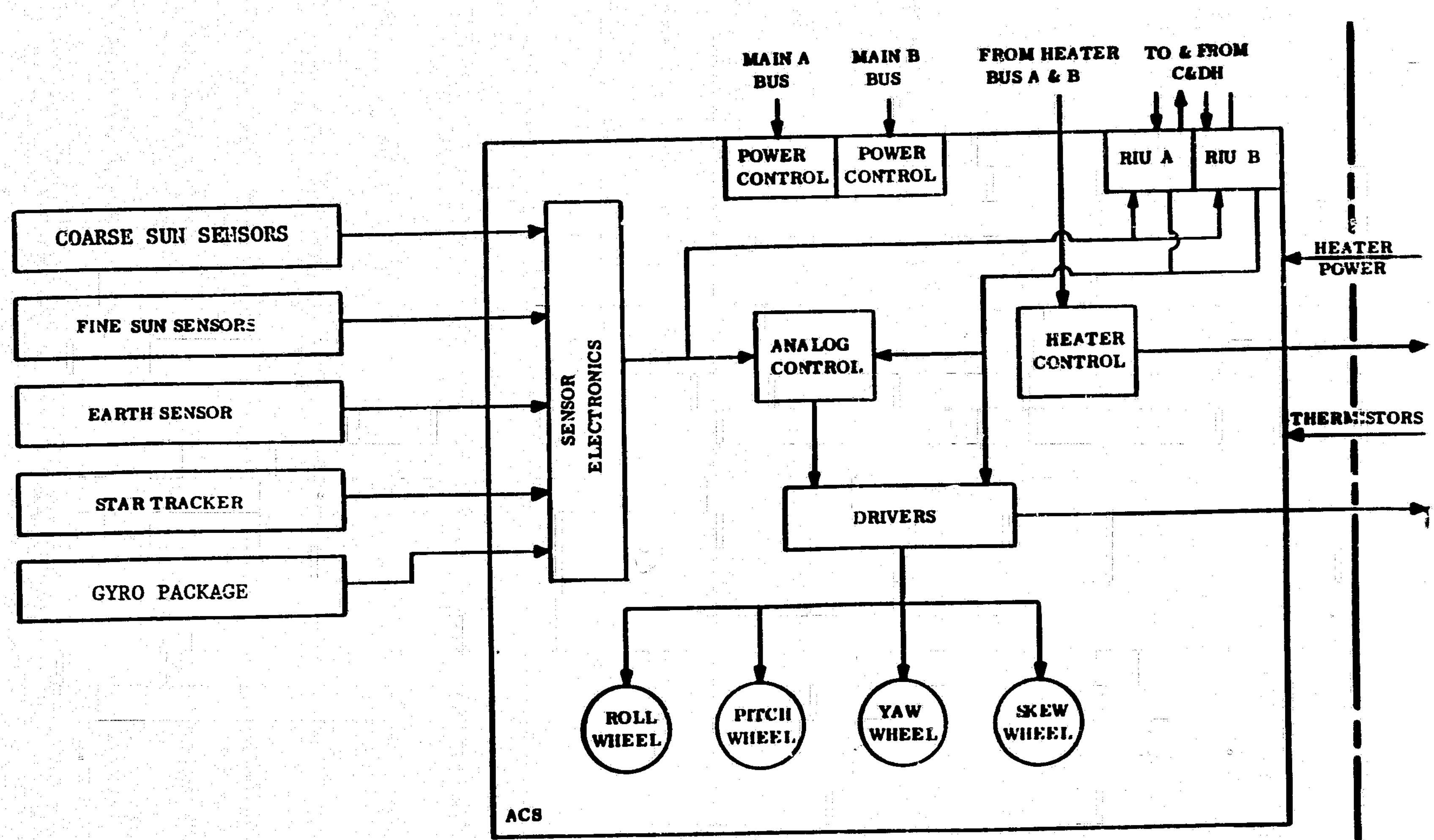


Figure 4.3-1. Altitude Control Subsystem

NOSS

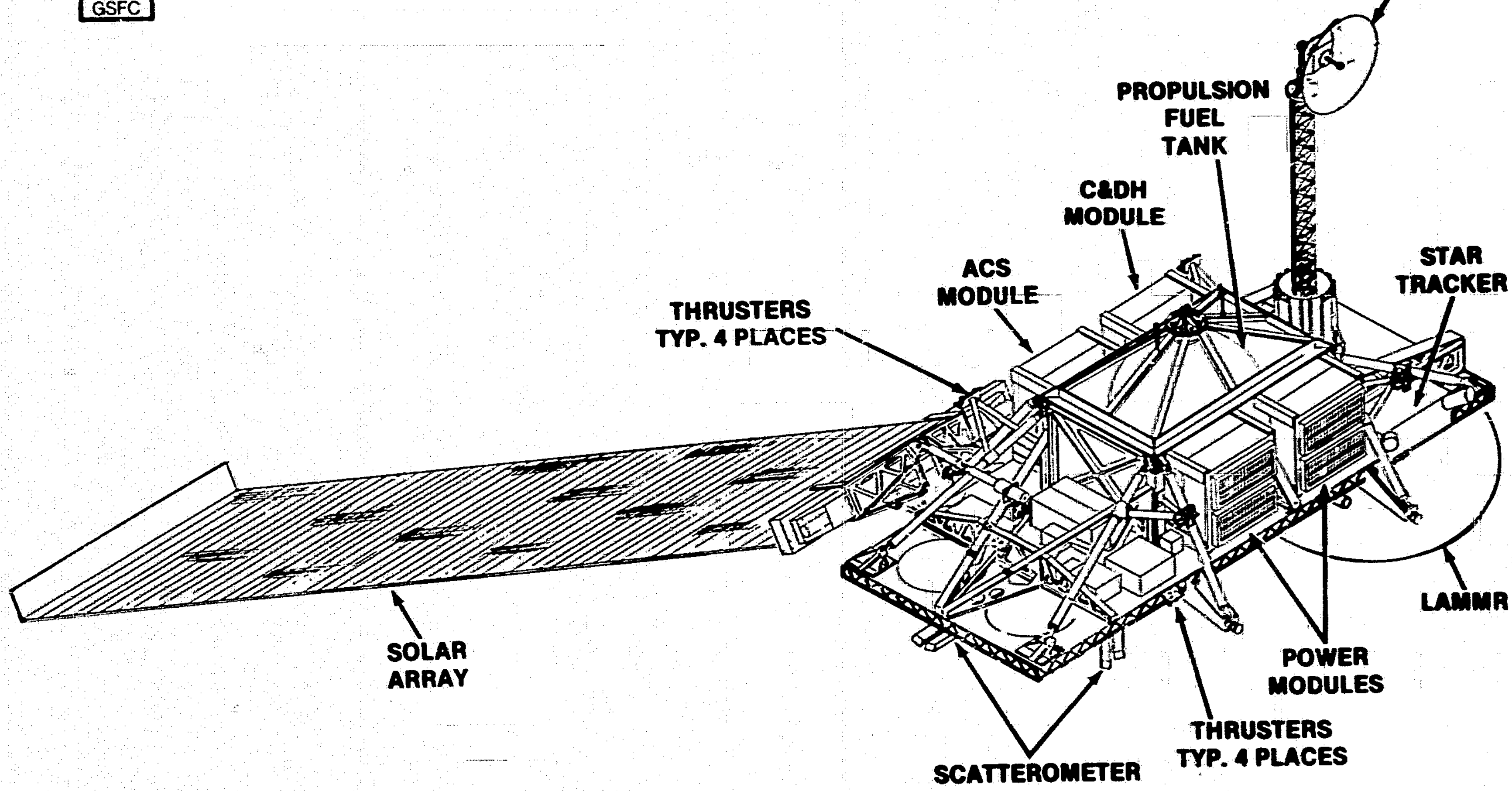


Figure 4.3-2. Space Viewing Side Of NOSS Spacecraft - On Orbit Configuration

3. Operational life of 3 years.

The NOSS attitude determination and orbit transfer requirements drive the ACS to be precise. An Inertial Reference Unit (IRU) with star trackers and a fine Sun sensor are required as detectors as well as a smooth torquing scheme for control.

The large size of NOSS results in large disturbance torques (on the order of 2×10^{-3} ft.-lb.) thus requiring a powerful torquing system for momentum control. Thrusters are used to unload wheels and are available for safe modes.

4.3.3 OPERATIONAL MODES

The ACS is designed to satisfy launch, deployment, acquisition, inertial-hold, orbit-transfer, safe, calibration, and retrieval modes of operation.

4.3.3.1 Launch Mode

The system is inactive during shuttle ascent except that the IRU gyros are powered and up to speed.

4.3.3.2 Deployment Mode

The system is capable of initializing the IRU using star tracker and Sun-sensor information while attached to the Orbiter Remote Manipulator System (RMS). The deployment sequence is discussed in Section 5 (Mission Timeline).

4.3.3.3 Acquisition Mode

Upon release by the orbiter RMS, the ACS goes into the acquisition mode. Reaction wheels are activated to reduce the spacecraft motion below an acceptable level and the thrusters are enabled to move the spacecraft away from the orbiter. The ACS orients the NOSS so that the array is normal to the Sun and the Earth center is in the +Z direction. At this time, an accurate

update of the IRU position is made and the system is placed in the inertial hold mode.

4.3.3.4 Inertial Hold Mode

This is the primary operating mode of the NOSS. The system is controlled by the Inertial Reference Unit with attitude updates from the FHST and FSS. Attitude position is maintained by the reaction wheels and/or the thrusters via information generated in the OBC which contains the system control laws.

4.3.3.5 Orbit Transfer Mode

Upon completion of spacecraft and experiment checkout, the ACS using the attitude control thrusters, orients the NOSS so that the +Z axis is into the velocity vector and the +X axis is Earth-center oriented as shown in Figure 4.3-3. The orbital transfer thrusters are then fired in a manner specified in Section 4.2. The orbital transfer thrusters maintain the +Z axis into the velocity vector while the attitude control thrusters control motion about the Z-axis with +X toward Earth.

When the spacecraft reaches the operational orbit, the spacecraft is reoriented with the Z-axis toward the Earth's center and the -X axis into the velocity vector as shown in Figure 4.3-3. The system switches into the inertial hold mode using reaction wheels with thrusters unloading for fine control.

4.3.3.6 Safe Modes

The system has separate safe modes utilizing electronics independent of those used in the normal operating mode. These electronics are analog and use coarse Sun sensors and/or the analog outputs of the FHST, IRU, FS, and ES.

The system is implemented such that the array is oriented normal to the Sun line while maintaining the X-Z plane aligned with the velocity vector and through the Earth's center. The mode allows for an array drive failure by having Sun sensors on both the array and the spacecraft. If the array drive

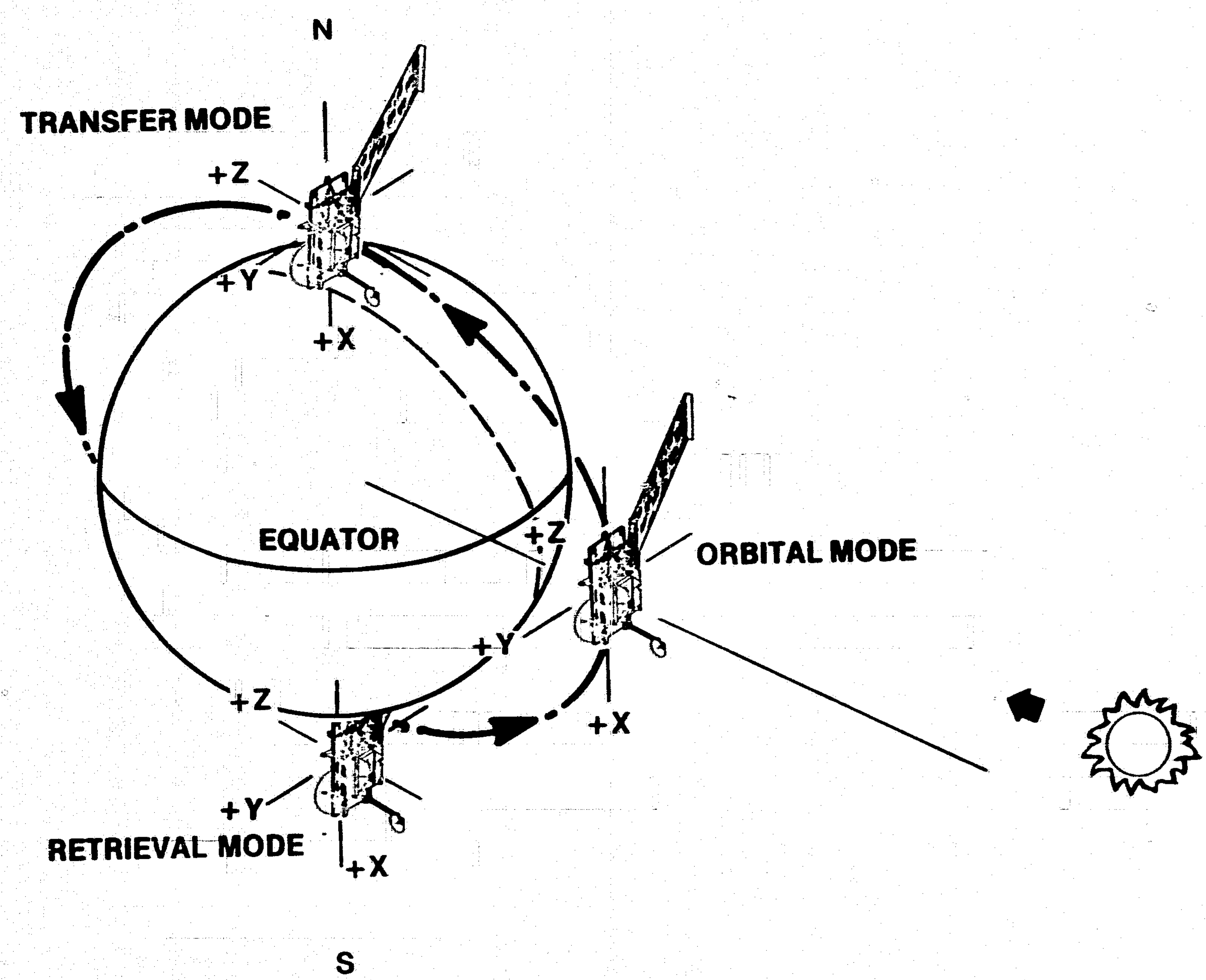


Figure 4.3-3. Spacecraft Orientation For Various Operational Modes

is operating, the spacecraft body is oriented with the +Z axis to Earth center and the -X axis into the velocity vector but with degraded performance.

4.3.3.7 Calibration Mode

Calibration of the IRU is normally done without taking the spacecraft out of the normal operating mode. Occasionally, it may be desirable to calibrate the star trackers by moving the spacecraft to selected positions. The star tracker and IRU data are then processed by the computer and the calibration data used for attitude determination.

4.3.3.8 Retrieval Mode

The retrieval mode is similar to the orbital transfer mode except that the spacecraft is oriented such that the -Z axis is into the velocity vector and the -X axis is toward Earth center. The large thrusters are fired to the orbit adjust and pulsed to control the -Z axis into the velocity vector and the control thrusters are used to keep the X-Z plane aligned to the Earth as shown in Figure 4.3-3.

When the required retrieval orbit is reached, the spacecraft is re-oriented onto its normal operating mode.

Contingencies must be determined and allowed for in the retrieval operation since the need to recover the spacecraft may be because of system failure.

4.3.4 EQUIPMENT DESCRIPTION

The overall attitude control system equipment is tabulated in Table 4.3-1. Components in the attitude control system are described in detail in the following paragraphs.

4.3.4.1 Inertial Reference Unit (IRU)

The IRU is the NASA Standard Inertial Reference Unit. The design insures that full functional capability of the ACS is maintained with the

TABLE 4.3-1

NOSS ATTITUDE CONTROL SUBSYSTEM EQUIPMENT LIST

Item	Quantity Number	Mass Kg	Power Watts	Size L x W x H (CM)	Telemetry Analog	Telemetry Digital	Command Pulse	Command Level	Command Serial
Sensors									
o Coarse Sun	10	.3	-	2 Dia x 1.25 L	-	-	-		
o Fine Sun	1	1.2	-	10 x 10 x 2.5	2	-	-		
o Earth	1	10.0	15	15 Dia x 40 L	4	-	2		
o Star Tracker	2	27.2	46	107 x 61 x 46	9	-	-		2
o Gyros	3	16.9	29.5	28 x 28 x 18	6	4	4		
Driver Mechanism									
Wheels and Drivers			20		8	4	8		
o Roll	1	10		35 Dia x 17 L					
o Pitch	1	10		35 Dia x 17 L					
o Yaw	1	10		35 Dia x 17 L					
o Skew	1	10		35 Dia x 17 L					
Attitude Control Electronics	1	25.2	35	(In Module)	40	18	42		4
RIU	2	4.3	5.0	6.5 x 20.6 x 18			1		
Structure	-	72.9	5.5	119 x 119 x 45.7 (Module)					
TOTAL		198.0	156.0		69	26	57		6

failure of any individual gyro or electronics channel. The gyro package is aligned to the spacecraft axes and located in the ACS module.

4.3.4.2 Fixed Head Star Tracker (FHST)

The FHST is the NASA standard fixed head star tracker. Each FHST includes a Bright Object Detector. The FHST's are located near the LAMMR as shown in Figure 4.3-2. The FHST and LAMMR (and other instruments as required) are aligned with respect to each other to insure meeting attitude determination requirements of the LAMMR.

4.3.4.3 Fine Sun Sensor (FSS)

The FSS is a two-axis precision device with a field-of-view of ± 30 degrees on each axis. The FSS is used to provide attitude updates to the IRU during times when there are infrequent stars available to the FHST and in event of a FHST failure. The NOSS unit is similar to the one being flown on the Solar Max Mission. The unit is aligned and calibrated with the FHST's and the IRU.

4.3.4.4 Coarse Sun Sensors

Eight Coarse Sun Sensors giving an effective field-of-view of 4π steradians are positioned on the spacecraft body. A stable null exists on the sunlight side of the spacecraft and both analog and computer modes of operation are available.

Additional coarse Sun sensors are located on the solar array and used to maintain the array normal to the Sun.

4.3.4.5 Earth Sensor (ES)

A CO_2-band Earth sensor utilizing a rotating mirror is mounted on the anti-Sun side of the spacecraft. It is capable of finding Earth nadir and holding the spacecraft on Earth nadir. It is used only during failure modes or as an additional source of attitude information.

4.3.4.6 **Reaction Wheels**

The reaction wheels are similar to the NASA Standard Reaction Wheels but have a torque capability of approximately 0.3 ft.lbs. at speeds between 0 and 50 percent of full speed (2500 rpm). The momentum storage capability is approximately 30 ft.lb. seconds.

4.3.4.7 **Attitude Control Electronics (ACE)**

Electronics to perform the following functions are packaged in the ACE in the ACS module:

- Power Conditioning and Switching
- Heater Control
- Control Logic and Timing
- Sensor Signal Conditioning and Processing
- Command Signal Processing and Switching
- Drive Signal Conditioning and Switching for Wheels and Thrusters
- Analog signal processing from sensor input to torque output
- Processing of ACS data.

C-2

4.4 COMMAND AND DATA HANDLING

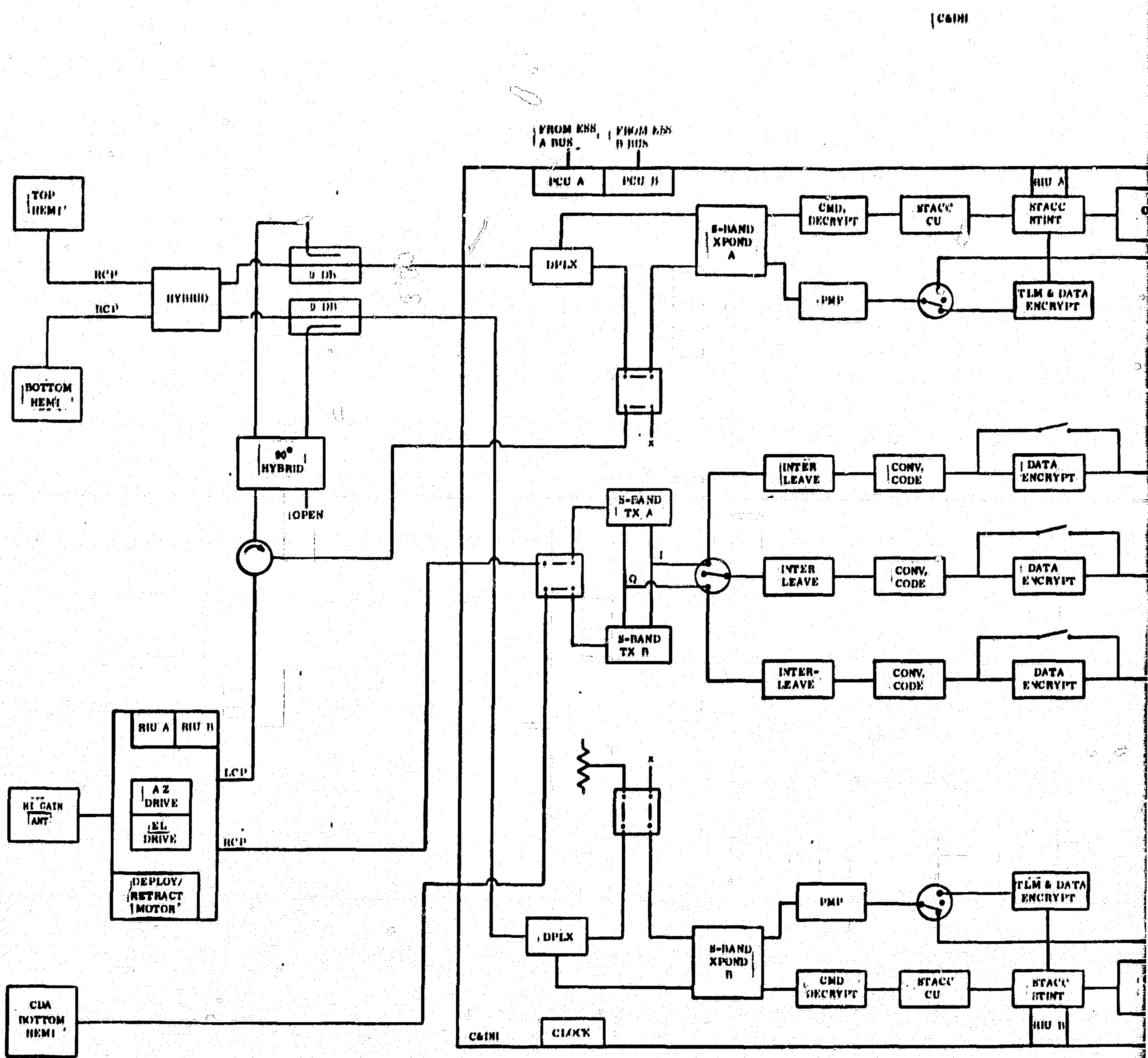
4.4.1 GENERAL DESCRIPTION

The NOSS Mission C&DH Subsystem consists of two distinct sections. First are those elements necessary to support the basic spacecraft subsystems and second a separate set of components necessary to satisfy functional requirements for the expected instrument complement. The C&DH configuration proposed provides the elements necessary for the spacecraft subsystems using existing equipment of proven design with instrument requirements being satisfied by equipment of new design. In both cases equipment redundancy has been implemented to achieve the five-year operation guideline.

4.4.1.1 Spacecraft C&DH Design

The MMS C&DH module provides the necessary core capability required by the NOSS mission. This equipment was selected since it provides an integrated subsystem of proven design. Figure 4.4-1 is a block diagram of proposed C&DH equipment and the subsystem equipment list is shown in Table 4.4.1. The NSSC-1 computer provides the processing necessary for the attitude control system, stores commands for delayed execution, monitors the spacecraft health and safety and can be used to distribute data collected from remote instruments, and spacecraft equipment. NOSS computer requirements were not sufficiently defined during the study to demonstrate that the NSSC-1 computer is compatible with NOSS spacecraft and instrument requirements.

The use of a data bus and remote interface units (RIU) provide the isolation necessary for successful command and data transfer on a spacecraft the size of NOSS. Each instrument and spacecraft subsystem is supplied with commands and necessary data transfer through RIU's located within its equipment.



FOLDOUT FRAME

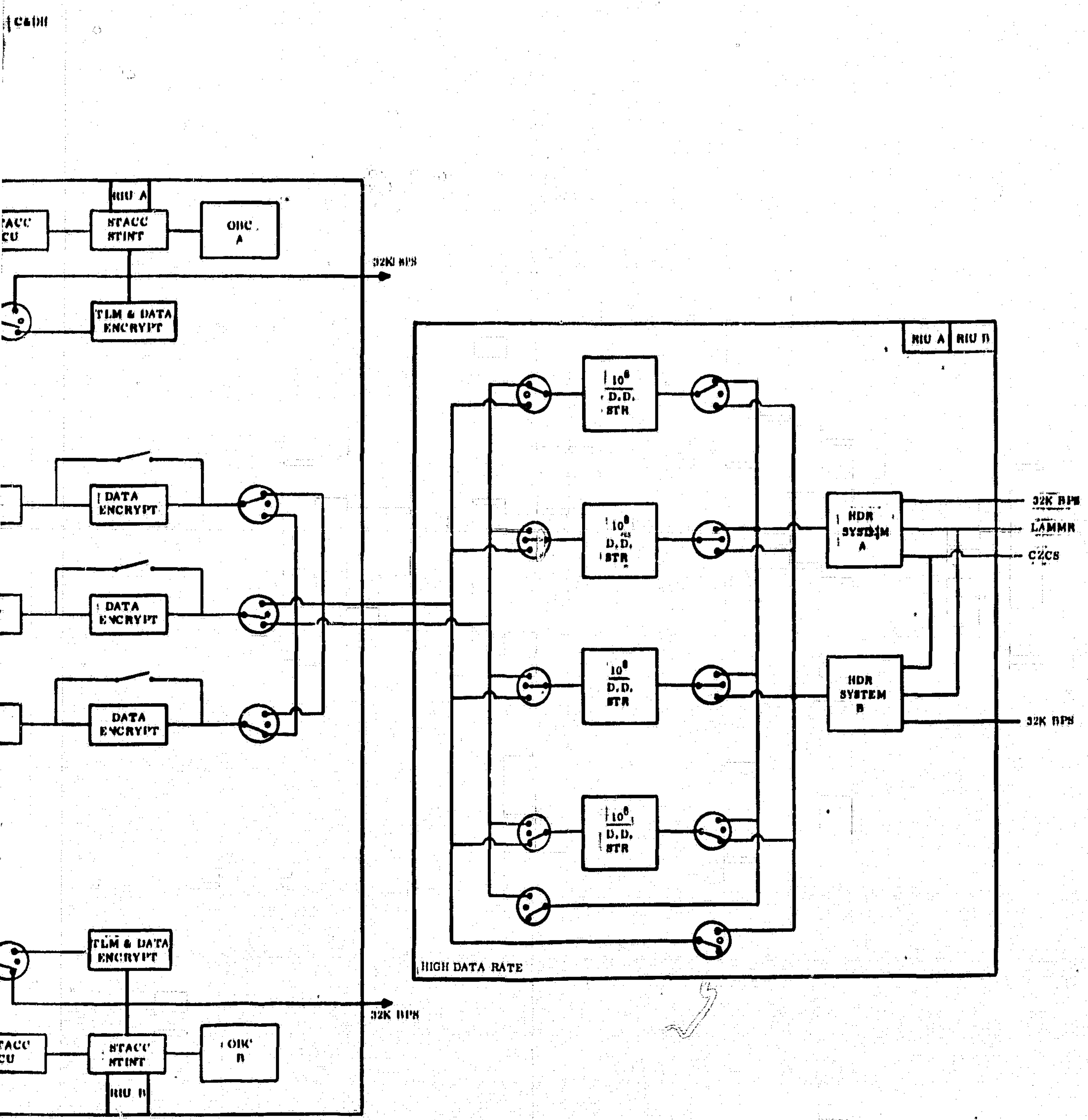


Figure 4.4-1. C&DH Block Diagram

FOLDOUT FRAME 2

TABLE 4.4-1

NOSS TELEMETRY, COMMAND AND DATA HANDLING SUBSYSTEM EQUIPMENT LIST

Item	Quantity Number	Mass Kg	Power Watts	Size L x W x H (CM)	Telemetry		Command		
					Analog	Digital	Pulse	Level	Serial
Communication and Data Handling Module	1	157.3	100	119 x 119 x 45.7	34	90	56	-	8
o On-Board Computer	2								
o STACC STINT	2								
o STACC CU	2								
o STACC RIU	2								
o PMP	1								
o RF Switches	2								
o Diplexers	2								
o S-Band Transponder	2								
o Clock	1								
o Switches	2								
o PCU	2								
Encrypter/Decrypter*	1	9.5	20	(in module)	--	--	--	-	--
o Telemetry and Data Encrypter	2								
o Command Decrypter	2								
o Data Encrypter	3								
o Conv. Encorder	3								
o Interleaver	3								
o S-Band Transmitter	2								
o RF Switches	1								
o Switches	7								

*In C&DH Module

TABLE 4.4-1 (cont.)

NOSS TELEMETRY, COMMAND AND DATA HANDLING SUBSYSTEM EQUIPMENT LIST

Item	Quantity Number	Mass Kg	Power Watts	Size L x W x H (CM)	Telemetry		Command		
					Analog	Digital	Pulse	Level	Serial
High Data Rate Box	1	56.8	48	38 x 45.7 x 30.5	24	8	32	–	4
o RIU	2								
o Tape Recorder	4								
o Hardware Box	2								
o Switches	10								
HGAS	1	88.6	40	130(D) x 53.36	6	40	14	–	2
OMNI Antennas	4	10.0							
o TDRSS	2								
o CDA	1								
o GPS	1								
GPS	1	17.3	35	31 x 42 x 21	10	32	12	–	1
o DPU									
o RPA									
TOTAL		339.5	243		74	170	114		15

The RIU receives coded data from a common data bus and develops commands or collects data. Commands of three distinct types are generated. These are discrete or logic level commands, relay drive commands, and 16-bit serial digital instructions. Data collected by the RIU can be of several types. Analog information is converted to 8-bit digital words. One milliampere constant current source is available for passive thermistor or transducer excitation and subsequent A/D conversion and digital bilevel data or serial 8-bit words can also be collected.

The design of the NOSS C&DH system including added equipment of new or existing design is fully redundant and operates from redundant essential power buses.

4.4.1.2 Instruments C&DH Design

The NOSS mission will use instruments of existing design with interfaces and data requirements that cannot be directly handled by the MMS C&DH equipment. Table 4.4-2 lists the data rates and command requirements for the four NOSS instruments. The science data from each instrument is available in a serial bit stream; however, the word structure is not common for each instrument. The altimeter (ALT) and scatterometer (SCAT) generate 10-bit words, the LAMMR is expected to have 12- or 14-bit words and the CZCS output is a parallel data transfer clock and 8 data lines with control and timing. Each instrument also requires some additional interfaces to the RIU to provide the needed temperature and housekeeping data. Compatible C&DH instrument interfaces will be provided in an Instrument Interface box (IFB) of unique design for each instrument.

Three independent data streams are generated for NOSS. Table 4.4-3 lists the contents and bit rate for each bit stream. Two of the three bit streams are stored onboard for high rate dump to TDRSS. These two recorded bit streams, 128 kbps and 1.2 Mbps, cannot be accommodated by the MMS C&DH equipment and the high data rate (HDR) equipment required is a new design.

Table 4.4-2

NOSS INSTRUMENTS COMMAND AND DATA REQUIREMENTS

Instrument	Data Rate	Commands
CZCS	1.2 MBS	40 Level
LAMMR	64.0 KBS	30 Pulse
Altimeter	8.5 KBS	8 Level
		16 Pulse
Scatterometer	4.0 KBS	12 Level
		18 Pulse

Table 4.4-3

NOSS TELEMETRY DATA CHANNELS

32 KBS Real Time TDRS Multiple Access		1.2 MBS Tape Recorder Ch 1 TDRS SSA Dump 2.66 MBS		128 KBS Tape Recorder Ch 2 TDRS SSA Dump 2.66 MBS	
LAMMR	4.0 KBS	CZCS	1.2 MBS	Real Time Data	32 KBS
ALT	8.5 KBS	GPS	*	LAMMR Data	64 KBS
SCAT	4.0 KBS	Time	*	Inst Growth	32 KBS
Inst Growth	8.0 KBS	S/C Att	*		
S/C HK	4.0 KBS				
GPS	0.25 KBS				
Time	0.5 KBS				
S/C Att	0.5 KBS				

* Included in 1.2 MBS

The HDR equipment performs the following functions:

a) Generate a synchronous data stream with frame sync pattern
b) Insert header data (time, location and spacecraft attitude)
c) Collect and buffer instrument data
d) Merge instrument data into synchronous bit stream
e) Collect, process and format CZCS data

The ALT and SCAT science data are collected by the RIU's and contained in the low rate 32-kbps data stream. This data is available for real time transmission to TDRSS using MA service but is not lost if real time transmission is not utilized. The entire 32-kbps data stream is merged into the 128-kbps data stream and recorded on tape recorder (TR) channel two.

4.4.1.3 Data Storage Design

The NOSS spacecraft is required to store its data onboard and execute a high rate tape dump to TDRS using the SSA mode. This mode of operation is required since it is not feasible to have a dedicated SSA channel for NOSS. Table 4.4-4 develops the storage capacity required for the two data streams being generated.

The NOSS guideline was for tape recorder playback to TDRSS for 15 to 20 minutes of each orbit. Tape recorder channel one which is dedicated to the CZCS instrument determines the maximum storage requirement of 1.8×10^9 bits per orbit. To meet this requirement, modification of a dual transport 10^8 bit standard tape recorder is planned. The maximum playback rate of this design is 2.66 Mbps which provides an adequate dump time of 11.3 minutes. Four recorders are available which provide some redundancy since one recorder can store up to two orbits of data for channel two. The capability exists for real time transmission of either HDR channels.

Table 4.4-4

NOSS DATA STORAGE REQUIREMENTS

Tape Recorder Channel 1

Data Rate = 1.2 MBS

Orbit Time = 100 Minutes

CZCS Duty Cycle = 25%

Storage Bits/Orbit = 1.8×10^{9}

Tape Dump at 2.66 MBS = 11.3 Minutes

Tape Recorder Channel 2

Data Rate = 128 KBS

Orbit Time = 100 Minutes

Instrument Duty Cycle = 100%

Storage Bits/Orbit = 7.68×10^{8}

Tape Dump at 2.66 MBS = 4.8 Minutes

4.4.2 COMMUNICATIONS

The NOSS mission requires the use of secure communication links. The data protection systems which provide this security is GFE from DOD. In the event of downlink telemetry difficulty relief from this security requirement is expected. The data system design allows the protection systems to be bypassed by command. Command protection equipment cannot be bypassed.

TDRS provides the primary communication services for NOSS. Forward or command rates of 125 and 1000 bps using SSA service and the TDRS standard transponder in the C&DH subsystem are implemented. Return or telemetry links at data rates of 32 kbs and 2.66 Mbs using both MA and SSA services supply real time and tape recorder dump data. Figure 4.4-2 illustrates the TDRS communication services capable of using the available spacecraft antenna complement. Backup ground station communications are also indicated.

The forward link at 125 bps has satisfactory margin into the spacecraft omni antenna system using TDRS SSA service. To provide command assurance, one of the two transponders will always be maintained in the 125-bps operating mode. The command bit rate is a command-selectable function for the transponder and use of the higher bit rate will be limited to periods of high command traffic with the transponder mode being returned to 125 bps upon completion of the high rate command activity. The 1000 bps command rate requires the use of the spacecraft high gain antenna for satisfactory link margin. TDRS MA Service provides command capability for NOSS only through the spacecraft high gain antenna.

The 32 kbps is formatted and transmitted using standard MMS C&DH module equipment. The transponder output at a normal 5-watt level and the spacecraft high gain antenna provide satisfactory margin for TDRS MA service. Additional equipment has been added to the standard C&DH module to provide for the 2.66-Mbps tape recorder dump telemetry. This equipment consists of sufficient hardware to provide both I and Q channel modulation of a 10-watt S-band transmitter. This equipment using the spacecraft high gain antenna provide only a marginally acceptable link using TDRS SSA service.

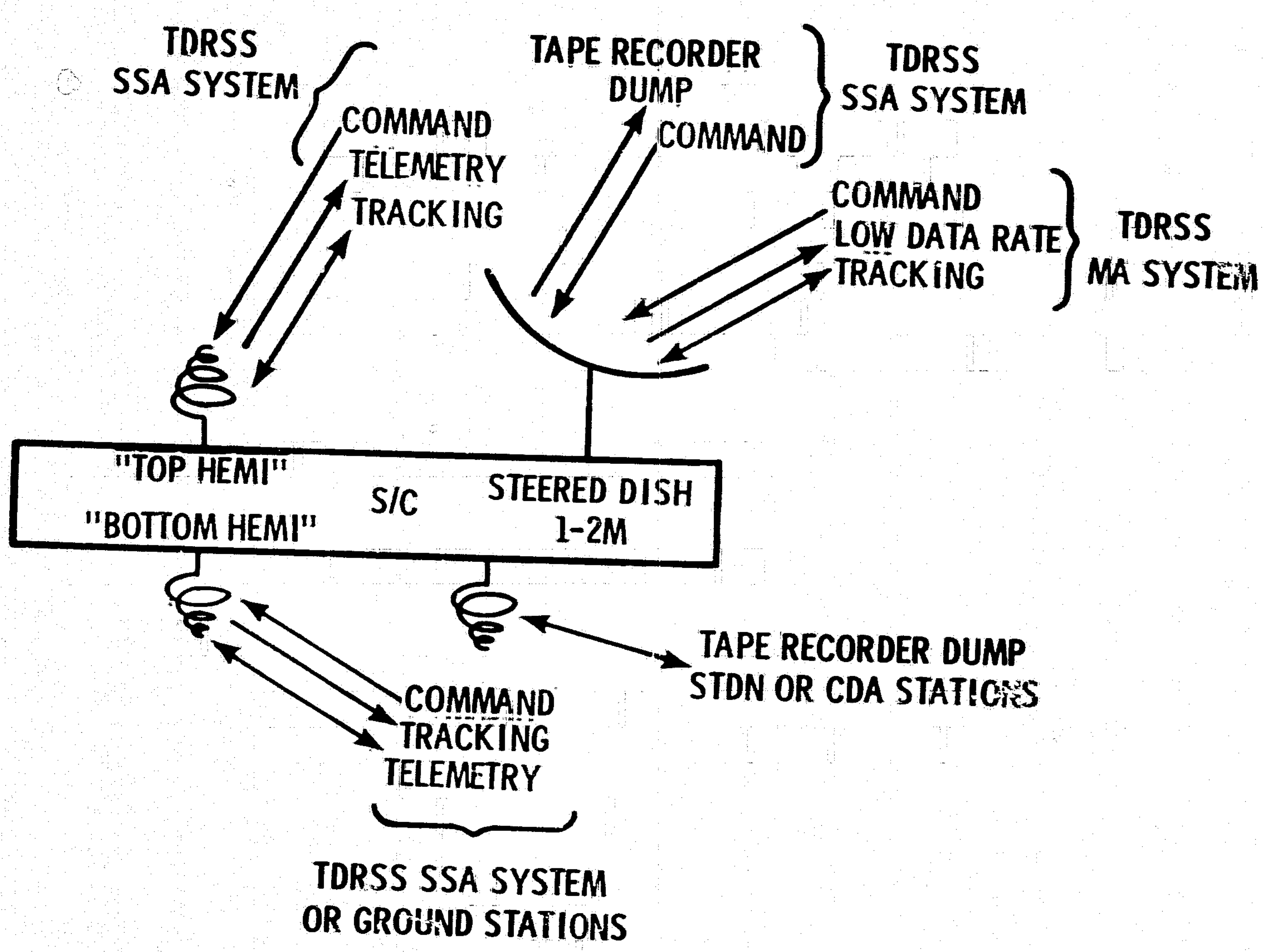


Figure 4.4-2 NOSS Communication Links

High rate tape recorder playback is required once per orbit. The analysis conducted to determine acceptable communication service has been based upon the TDRS users guide and the MMS users manual. There are no spacecraft missions presently planning to use TDRS SSA service at data rates exceeding 1.0 Mbps. Plans to implement a system at 2.66 Mbps should be confirmed by a request for formal TDRS acceptance.

The NOSS communication design provides for services which may be available from STDN or CDA ground stations. These services would be available on an intermittent basis and would be used in event of spacecraft malfunction or TDRSS problems.

4.5 MECHANICAL

4.5.1 SPACECRAFT DESCRIPTION

The on-orbit configuration of the NOSS satellite is shown in Figure 4.5-1 and 4.5-2 and combines major elements, a nadir-pointing instrument payload, MMS modules, and mission-unique hardware. The structure necessary to support these elements, antennas, and solar panels is also shown. The subsystem equipment list is presented in Table 4.5-1.

4.5.1.1 Configuration

The NOSS spacecraft is designed as an instrument platform and all instruments are located on the Earth-side of the spacecraft. The instrument complement consists of four core sensor packages and a possible experimental payload. Mission support packages, laser retroflector, and GPS are also part of the baseline. The design integrates the major elements to the spacecraft such that their physical requirements are satisfied, they are provided a benign environment in which to function, and they have compatible neighbors.

The NOSS design satisfies a complex set of spacecraft system, sensor and engineering subsystem requirements during launch, orbital, and retrieval operation phases of the mission. Designed as an instrument platform with horizontal orientation in the shuttle bay, NOSS provides a clear field-of-view for all instruments with a minimum of deployments. Additionally, the design provides a capability for a 25 percent growth in the instrument complement.

4.5.1.2 Propulsion Subsystem

The propulsion subsystem incorporates the Lockheed 1.93-m (76-inch) tank with 1724 kgm (3800 lb) of hydrazine mono-propellant. Being the single heaviest subsystem (more than 1993 kgm), the propellant tank and associated plumbing are located near the center of the spacecraft mid-way between the forward and aft longeron attach points and directly over the keel fitting. This configuration permits a minimum of plumbing to the thrusters. Since the spacecraft and propellant tank center-of-masses are aligned, center-of-mass shifts due to hydrazine propellant expenditure are minimized. The propellant

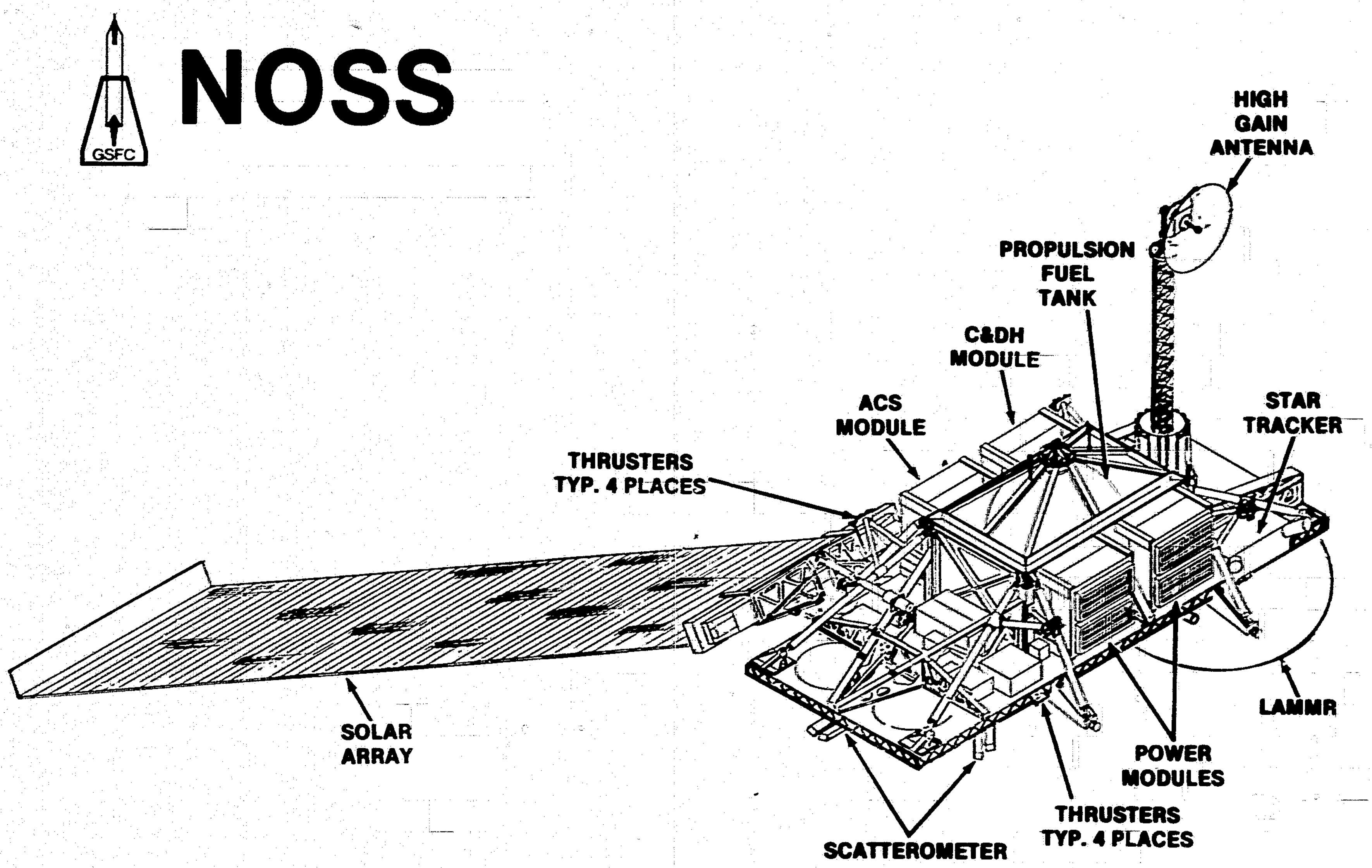


Figure 4.5-1. Space Viewing Side Of NOSS Spacecraft - On Orbit Configuration

NOSS

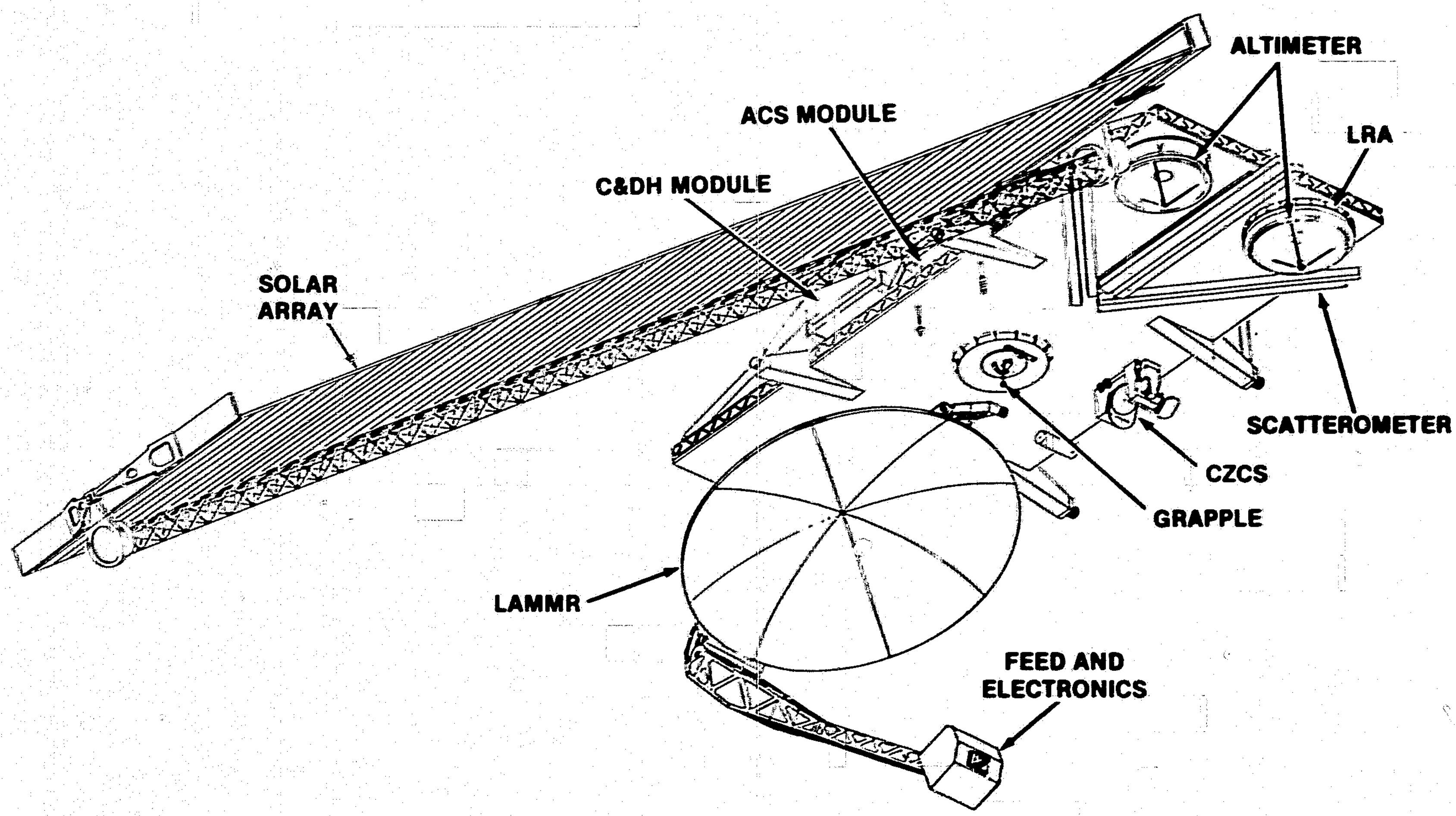


Figure 4.5-2. Earth Viewing Side Of NOSS Spacecraft - On Orbit Configuration

TABLE 4.5-1

NOSS MECHANICAL SUBSYSTEM LIST

	Mass KG	Power Watts	Telemetry		Command		
			Analog	Digital	Pulse	Level	Serial
Mechanism	23		4	-	8	-	-
Structure	1,142	-	-	-	-	-	-
Harness	159	-	-	-	-	-	-
Thermal							
Blankets	109	400	40	-	10	-	-
Attachments	27						
TOTAL	1,460	400	44		18		

tank and its associated plumbing can be integrated as a subsystem at the spacecraft level. Pressurant and propellant servicing is through fill and drain valves located on the side away from the instruments and electronics.

4.5.1.3 LAMMR

All elements of the spacecraft are stowable within the shuttle orbiter's prescribed dynamic envelope. The 3.6-m diameter LAMMR occupies a significant portion of the orbiter payload envelope in the stowed launch configuration. The LAMMR is located on the +X wing of the spacecraft. The LAMMR feed would violate the shuttle orbiter payload dynamic envelope if it were fixed and is, therefore, folded and latched with the arm parallel to the orbiter X-axis during launch and retrieval. Mechanical devices such as drive motors and gears are the primary mode for stowing prior to retrieval. As a backup for retrieval, EVA is recommended for LAMMR feed arm caging with a manual jettison capability as additional backup.

4.5.1.4 CZCS

The CZCS is positioned on the +Y axis near the edge of the instrument platform to provide an unobstructed Earth field-of-view for data collection. Unobstructed views of space are provided for instrument calibration and for the radiative cooler. The radiative cooler door is latched and the scan mirror is caged during launch.

Due to the potential for contamination of the CZCS in the shuttle bay, it is probable that a protective cover will have to be provided to cover the CZCS during launch. This housing is not depicted in the concept shown here.

4.5.1.5 Scatterometer

Six 3-m scatterometer (SCAT) antennas are positioned at the -X wing of the spacecraft, symmetrical about the X-axis and tilted slightly to provide the desired surface antenna pattern. The position provides a clear Earth field-of-view including sidelobes. Redundant electronic boxes for the

SCAT are located near the antennas within the spacecraft structure.

4.5.1.6 Altimeter

Two radar altimeters are positioned on the -X wing of the spacecraft and are positioned symmetrically about the X-axis in locations which allow the central portion of the platform to remain available for growth of the instrument complement. Cylindrical rf equipment sections are attached to the antenna base and rectangular signal processor sections are located nearby within the structure.

4.5.1.7 LRA

A toroidal laser retroflector array surrounds the altimeter on the +Y (Sun) side of the instrument platform.

4.5.1.8 Solar Array

A 3.86 by 14.2 m solar array with a single degree-of-freedom capability for looking directly at the Sun is attached to the primary structure on the -X wing of the spacecraft. This location provides an unobstructed pointing capability for the solar arrays to directly view the Sun and minimum interference with the required fields-of-view of the NOSS instruments and antennas. The solar array subsystem consists of the modified STS Power Extension Package (PEP) and a BBRC drive system. The array is a fold-out design with a deployable lattice-boom deployment/retrieval scheme.

During launch and retrieval the solar array is stowed and caged and positioned parallel to the Y-axis, across the orbiter cargo bay, and within the orbiter's dynamic envelope.

After spacecraft removal from the shuttle bay, the solar array is deployed by the PEP and BBRC drive system. Prior to retrieval, the solar array will be stowed and latched to the spacecraft structure. Jettison and/or EVA should be provided in the event of failure of the stowing or latching mechanisms.

4.5.1.9 **Sun Sensors**

Coarse Sun sensors are located on the solar array.

4.5.1.10 **Low-Gain Antennas**

Two low-gain antennas, each providing hemispherical coverage, are located on the spacecraft. The antennas are fixed-mounted, nominally zenith and nadir pointing. The zenith-oriented LGA is mounted aft and near the keel fitting. The nadir-oriented LGA is located near the middle of the instrument platform on the Earth-side of the spacecraft.

4.5.1.11 **Command and Data Acquisition Antenna**

A nadir-oriented Command and Data Acquisition (CDA) antenna located on the instrument platform provides hemispherical coverage.

4.5.1.12 **Earth Sensor**

An Earth sensor is located on the +Y axis of instrument platform near the CZCS.

4.5.1.13 **Global Positioning System**

A zenith-oriented GPS L-band antenna is fixed-mounted on the space side of the spacecraft forward of the keel fitting. GPS electronics are mounted within the spacecraft structure.

4.5.1.14 **High-Gain Antenna**

A 1.8m two degree-of-freedom HGA is located on the space side of the spacecraft in a position to view one TDRSS for the majority of the orbit without blockage from any spacecraft appendage.

The antenna is SMM high-gain antenna system with deployment, retrieval, caging and jettison capabilities. The canister has been lengthened

to 1.19 m to accommodate the additional deployable lattice-boom turns required for the increased deployment length. The antenna deploys 2.50 m from its stowed position. The canister attachment scheme has been modified for NOSS and the deployed antenna is 4.14 m from the canister attachment point.

4.5.1.15 Star Trackers

Star trackers are located on the shady side of the spacecraft and are oriented to provide zenith coverage during orbital and transfer modes. The star trackers are oriented perpendicular to one another symmetrically about the Y-axis and are inclined 30° to space from the instrument platform. Star trackers are positioned in proximity to the LAMMR to provide accurate LAMMR pointing data.

4.5.1.16 MMS Modules

The spacecraft design is based on the use of four standard MMS modules with minimum modifications clustered about the centrally-located propellant tank. Since power requirements exceed the capabilities of a single MMS power module, two MMS power modules are provided; both are located on the shady side (-Y side). An attitude control system module and a command and data handling module are located on the Sun-side. All modules are standard MMS and mounted in standard MMS fashion.

4.5.1.17 Instrument Electronics

Instrument electronics are generally grouped toward the -X end of the spacecraft while the spacecraft-related electronics are located toward the aft end. Packages are thermally isolated and arranged to meet harnessing and mass properties requirements.

4.5.2 SPACECRAFT STRUCTURE DESCRIPTION

The NOSS structure provides structural integrity during all phases of the mission including shuttle landing loads. Interfaces for shuttle bay restraint for launch and retrieval are also provided. The structure provides

support and attachment points for the modules, support for the instrument platform, attachments for the solar array, and support for the propellant tank and thrusters. Figures 4.5-3 and 4.5-4 show the spacecraft in the stowed configuration. Mechanical devices are used to unlatch, deploy, stow, and relatch equipment that is attached to deployed booms and articulating members.

The LAMMR feed (Figure 4.5-3) is folded and latched to the LAMMR structure to fit within the orbiter payload dynamic envelope. Figure 4.5-4 shows the solar array tied down and locked to the support structure for ground handling, launch, and retrieval. The arrays are structurally attached to a boom support structure and the boom is attached to the spacecraft structure.

The structural backbone of the spacecraft is provided by two lateral beam structures which connect the longeron retention points. The longeron retention points interface with the orbiter at stations 1010 and 1128. Figure 4.5-5 shows the central support structure which joins and reinforces the lateral beam structures and provides the keel retention point. The NOSS keel retention point interfaces with the orbiter at station 1069. Figure 4.5-6 shows the shady side of the spacecraft, Figure 4.5-7 shows the Sun side, and Figure 4.5-8 shows the forward end.

The central support structure is a box with shear panel lateral sides. Shear panels extend radially inward to a cylindrical center tube which supports the 1.93-m (76-inch) diameter propellant tank. A pyramidal structure joins the box to the keel trunnion. The central support structure reinforces the platform extensions via truss structure and cylindrical struts.

The platform consists of an external frame with channel intrastructure and honeycomb shear panels. The intrastructure provides local interface attachment and reinforcement for the heavier instruments. The honeycomb panels provide interface attachment for lighter instruments and antennas. Figure 4.5-9 shows the layout of instruments and antennas on the platform.

Local structure is provided for support of the solar array subsystem, modules, high-gain antenna, thrusters, and antennas. The primary structural members are aluminum and conventional assembly techniques are used.

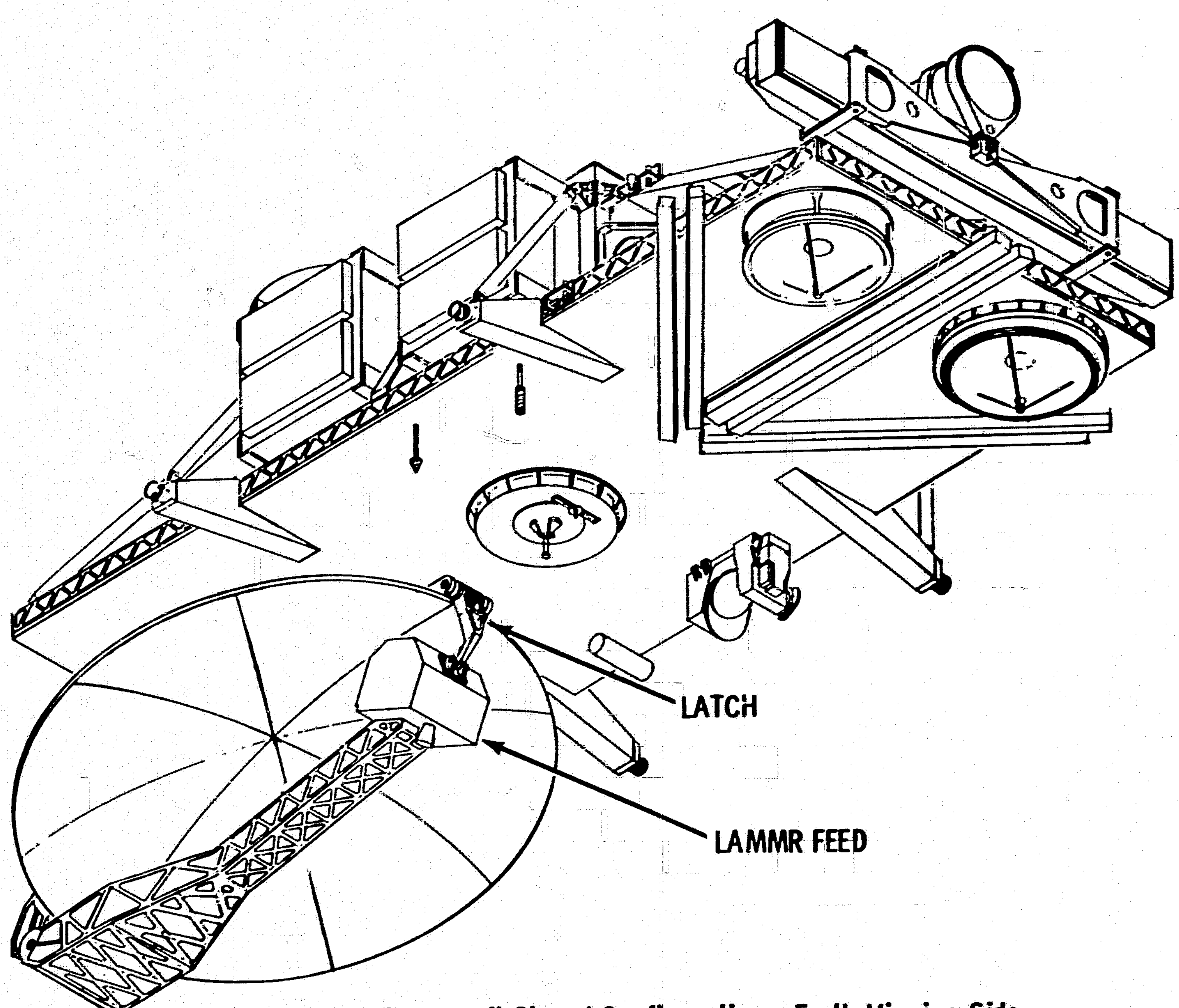


Figure 4.5-3. NOSS Spacecraft Stowed Configuration - Earth Viewing Side

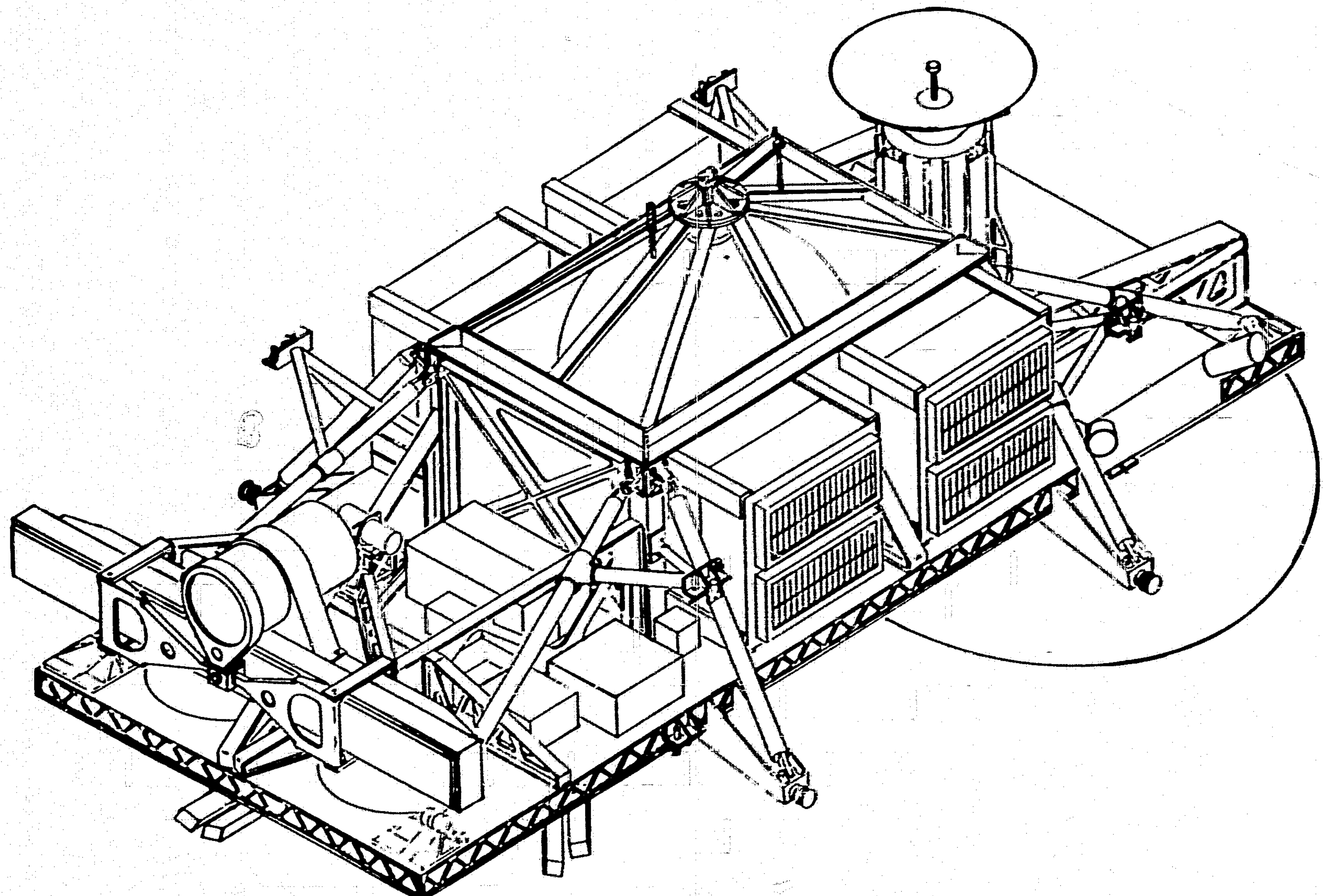


Figure 4.5-4. NOSS Spacecraft Stowed Configuration - Space Viewing Side

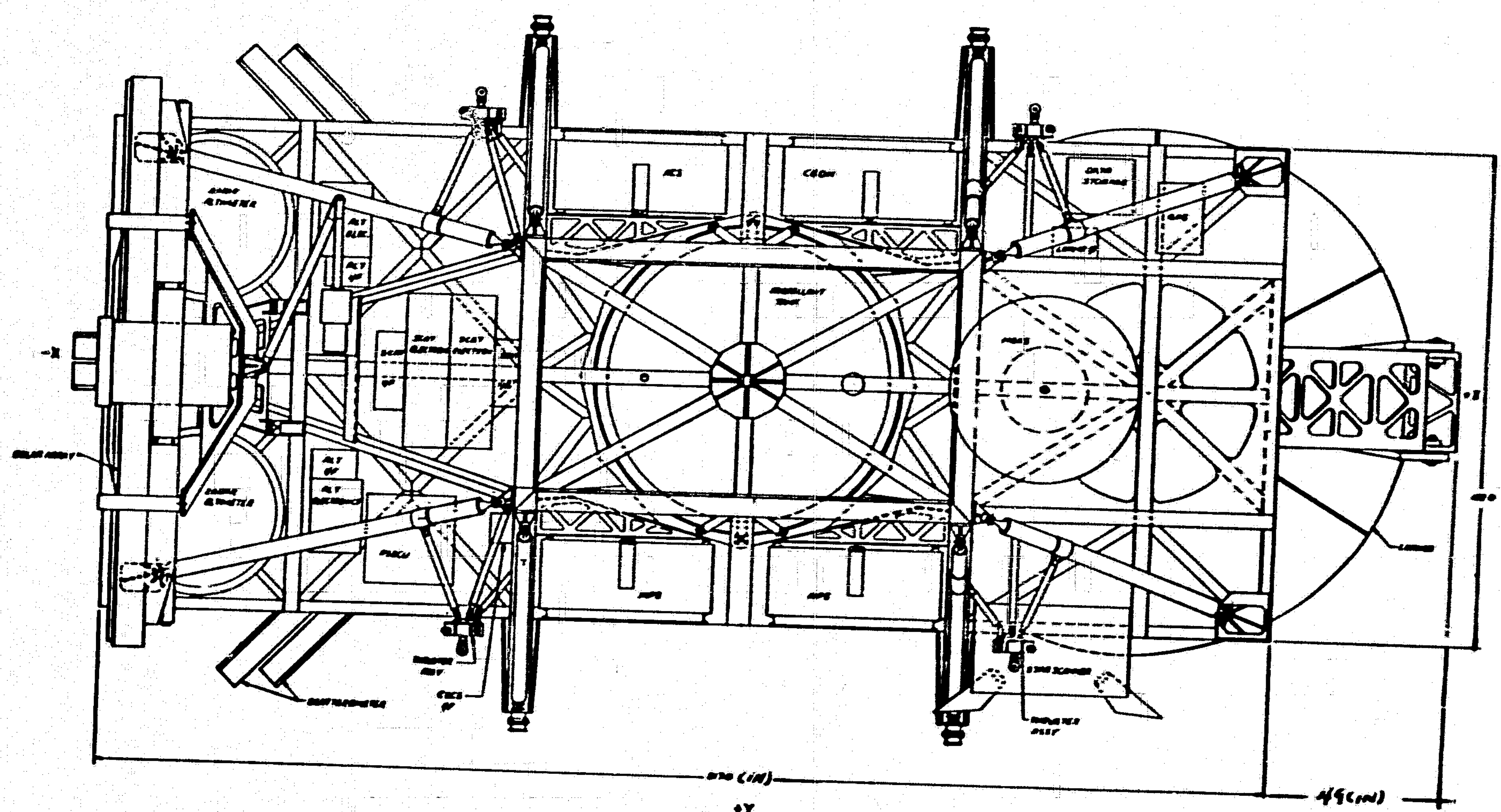


Figure 4.5-5. NOSS Spacecraft Space Viewing Side - Stowed

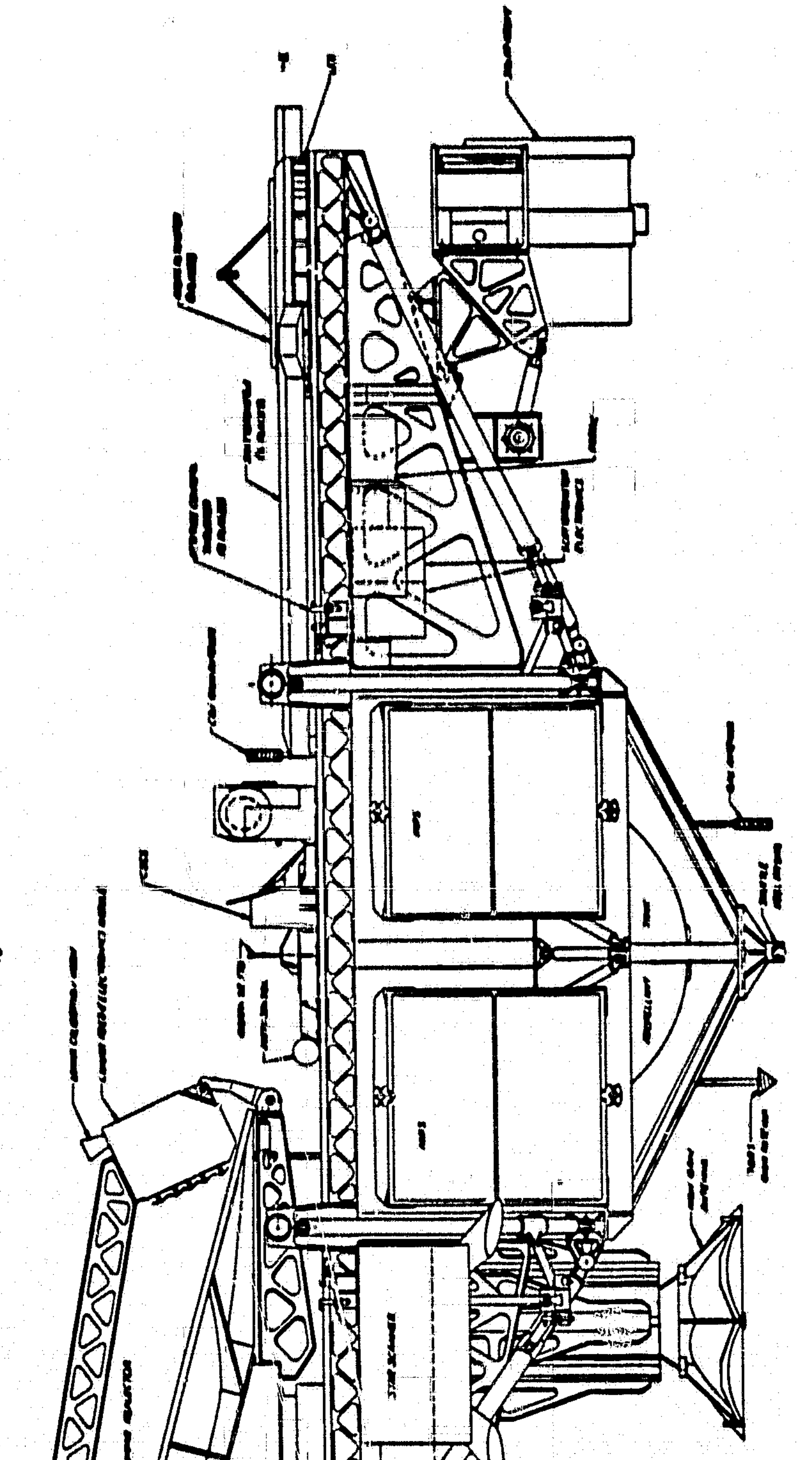


Figure 4.5-6. NOSS Spacecraft Side View - Cold Side

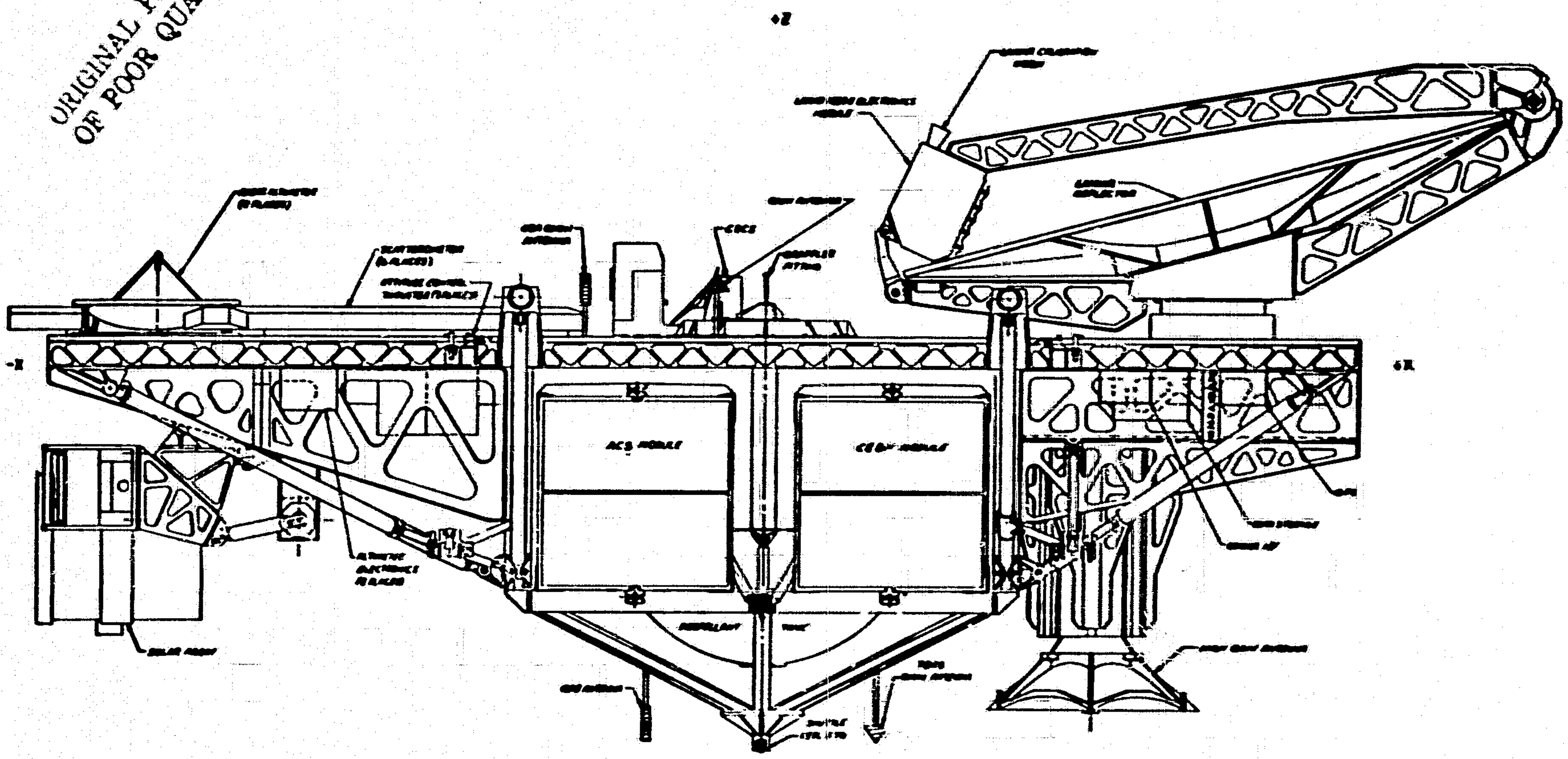


Figure 4.5-7. NOSS Spacecraft - Side View - Sun Side

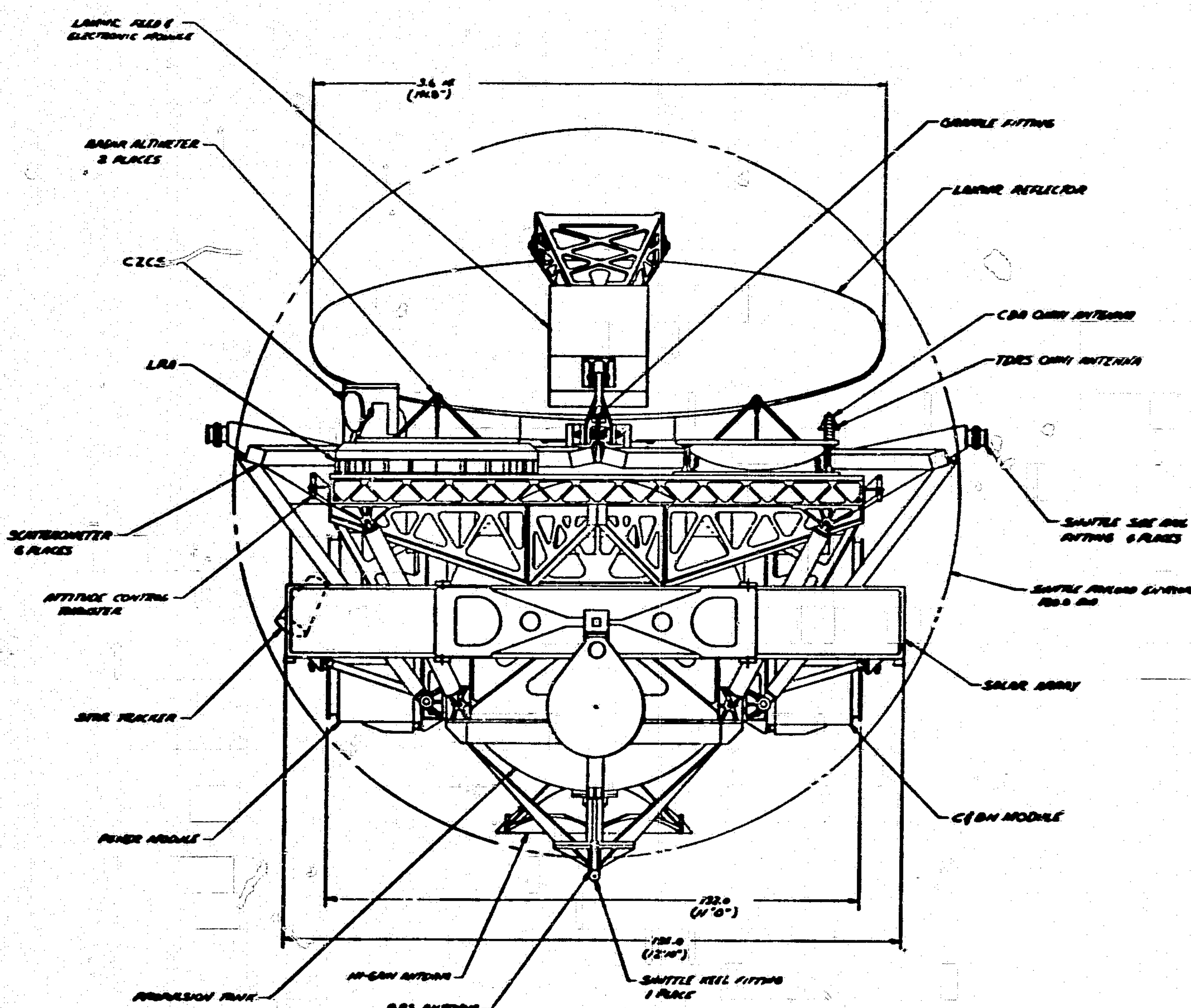


Figure 4.5-8. NOSS Spacecraft - End View Forward End

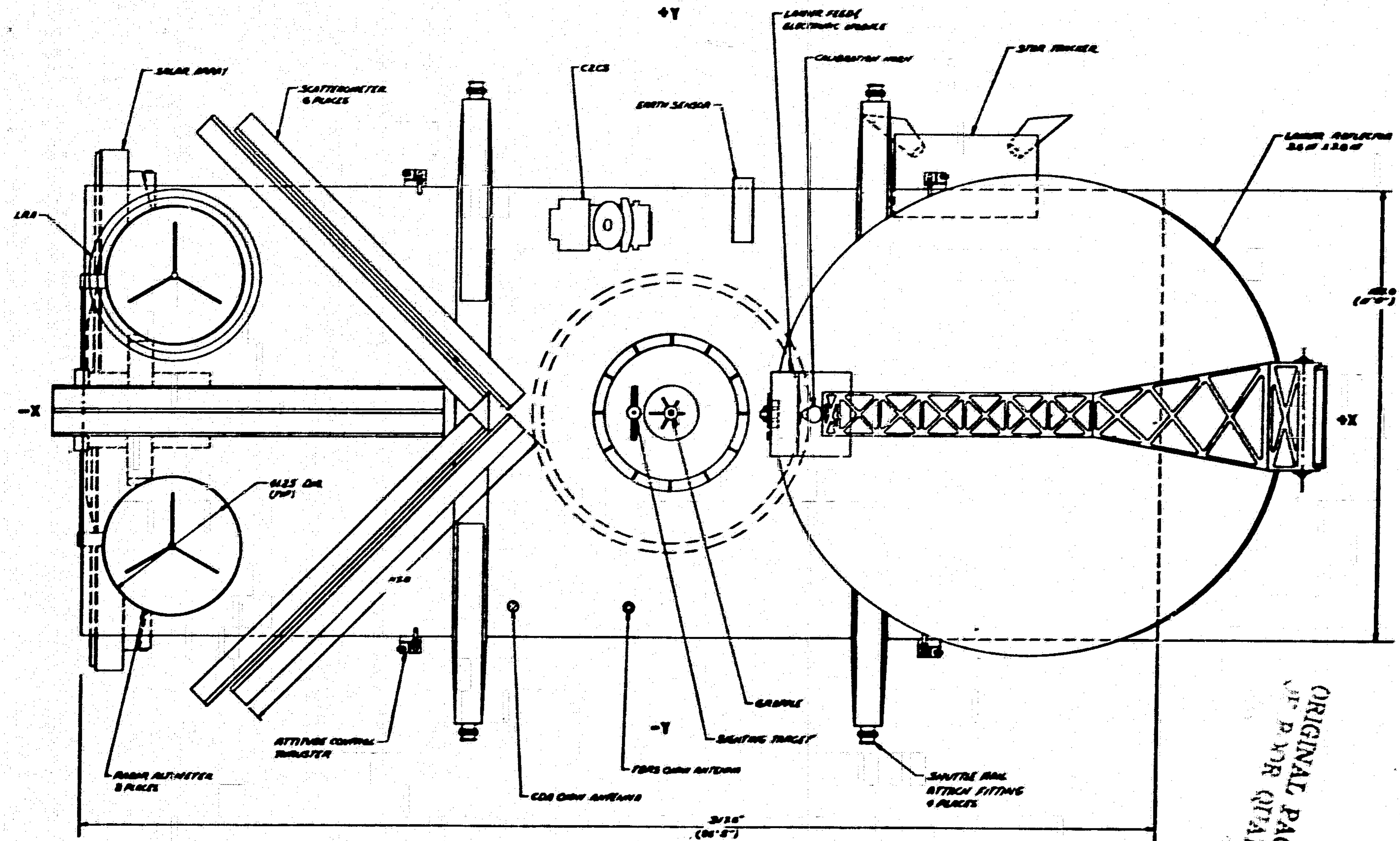


Figure 4.5-9. NOSS Spacecraft - Earth Viewing Side - Stowed

Although a detailed structural analysis was not conducted to size NOSS structural members, a preliminary analysis of trunnion-orbiter interface loads shows an adequate factor of safety.

4.5.3 MASS PROPERTIES SUMMARY

This preliminary mass properties analysis was completed early in the study and was not updated. They are included here to give representative numbers for C.G. position and moments of inertia. The current mass of the spacecraft is summarized in Table 2.5-1.

Major elements of the NOSS are arranged to minimize center-of-mass shifts due to mass expulsion, appendage deployment, or articulation. Major components are arranged such that the NOSS principal axes coincide with control axes. Analysis has not been performed to confirm that these goals have been met and changes in component location may be required. Major elements have been arranged such that the NOSS center of mass is aligned within prescribed limits for the orbiter payload bay and is discussed further in Section 4.5.4.

No attempt has been made to optimize or reduce weight. NOSS mass distribution is shown in Figure 4.5-10. Instrument electronics are grouped toward the -X end of the spacecraft while spacecraft-related electronics are located toward the aft end. Based on this mass distribution, and including a 25 percent instrument growth factor plus a 15 percent contingency, the NOSS launch mass is 6015 kgm (13233 lb). After expending 1455 kgm (3200 lb) of propellant the mass at retrieval is 4560 kgm (10033 lb.).

Mass properties of 3.6-m (12-ft) and 4.6-m (15-ft) wide spacecraft are shown in Table 4.5-2. The 3.6-m wide NOSS has the following mass properties at launch and retrieval:

		Launch	Retrieval
Mass (kgm)		6015	4560
C.M.	$\overline{X}$ (cm)	17.8	23.6
	$\overline{Y}$ (cm)	0	0
	$\overline{Z}$ (cm)	41.9	24.4

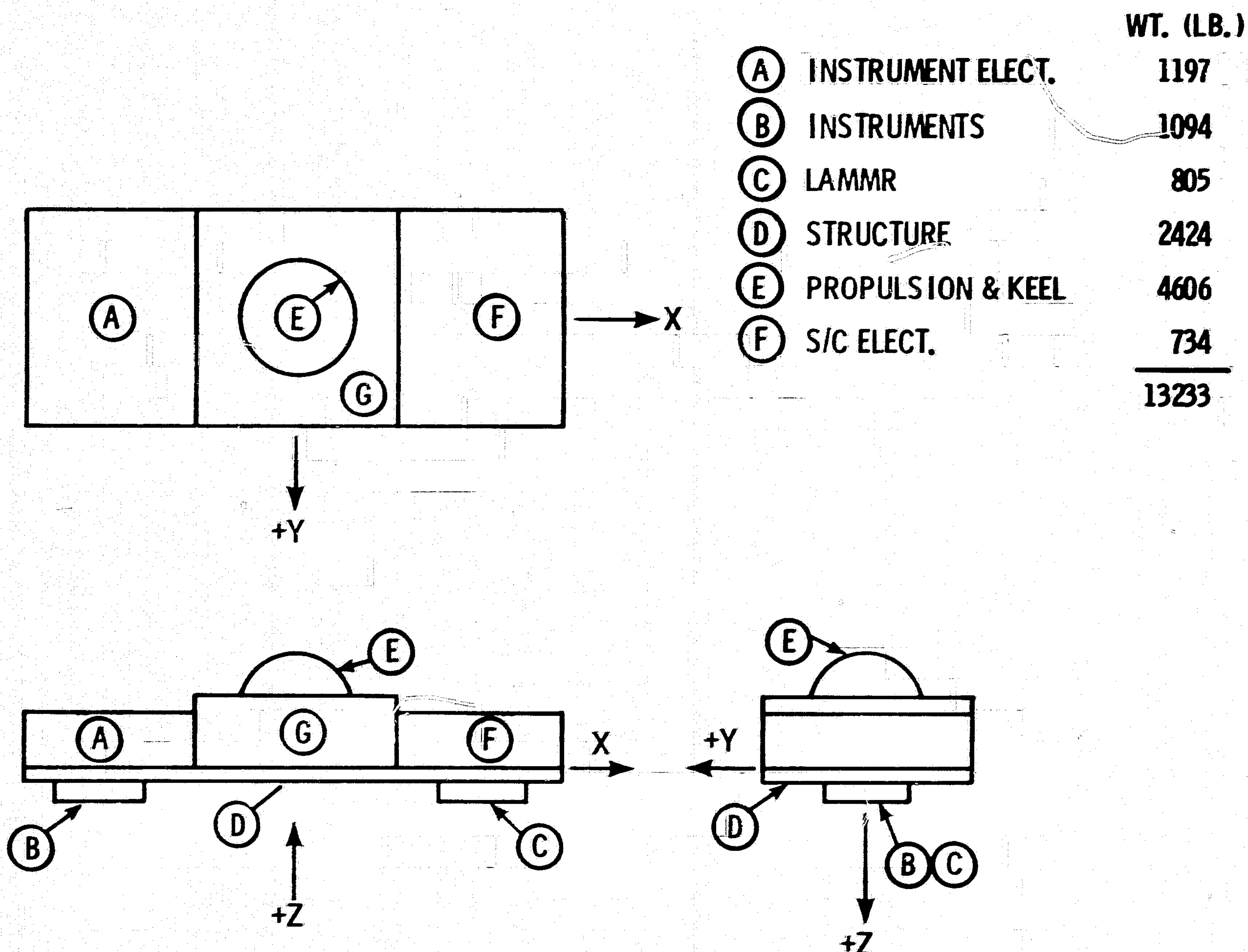


Figure 4.5-10. NOSS Weight Distribution

TABLE 4.5-2

MASS PROPERTIES SUMMARY

PROPERTY \ CASE		12 ft Wide S/C		15 ft Wide S/C	
		3200 lb Propellant	Dry Weight	3200 lb Propellant	Dry Weight
Weight (lbs)		13,233	10,033	13,233	10,033
I_{XX} (lb-in^2)		20.8	17.2	26.8	23.2
I_{YY} (lb-in^2)		99.2	95.3	99.2	95.3
I_{ZZ} (lb-in^2)		104.7	102.8	110.7	108.8
I_{XZ} (lb-in^2)		1.1	0.5	1.1	0.5
** $I_{XY} = I_{YZ}$		± 1.0	± 0.5	± 1.0	± 0.5
SYSTEM C.G. LOCATION	$\overline{X}$ (in)	- 7.0	- 9.3	- 7.0	- 9.3
	$\overline{Y}$ (in)	0.0	0.0	0.0	0.0
	$\overline{Z}$ (in)	- 16.5	- 9.6	- 16.5	- 9.6

* All mass moments of inertia are to be multiplied by 10^6 and are about the S/C center of mass.

** Assumed values for I_{XY} and I_{YZ}.

4.5.4 NOSS/STS STRUCTURAL INTERFACES

4.5.4.1 Spacecraft Location in Orbiter

The spacecraft is positioned in the orbiter payload bay as shown in Figure 4.5-11. The aft payload position was selected to utilize the greater load capacity in that area of the orbiter and to provide adequate RMS accessibility. Standard STS retention fittings and locations were used for NOSS spacecraft.

The side trunnions are attached at longeron retention points 1010 and 1128 and keel retention point is 1069 (see Figure 4.5-12). Spacecraft mass is distributed about these attachments. The origin of the X-axis is at station 1069, midway between the forward and aft trunnions. The most massive single element of the spacecraft, the propellant tank, is also centered at 1069. The origin of the Y-axis is midway between the longerons used for the side attachments and at the center of the payload envelope. The origin of the Z-axis is on the space viewing side of the platform and 3 inches below the center of the orbiter payload envelope.

The NOSS coordinate axes are the same as the orbiter axes with the LAMMR to the rear.

The design maintains a greater than 4-inch clearance on the 180-inch diameter payload envelope. (See Figure 4.5-8).

4.5.4.2 Structural Loading Arrangement

A five-point payload retention system (indeterminate) (Figure 4.5-13) was selected for the NOSS spacecraft. Stabilizing fittings at the forward location (1010) react Z-axis loads only, X-axis and Z-axis loads are reacted in the primary fittings at the aft location (1128) and the Y-axis loads are reacted in the keel fitting. A preliminary analysis of trunnion limit loads, assuming a rigid orbiter, was conducted to verify NOSS conformance with orbiter-payload interfaces and requirements. This preliminary static analysis showed a 2.5 minimum factor of safety based on loads defined in the JSC Volume XIV. A flight loads analysis is required to analyze the indeterminate configuration and to size the structural members.

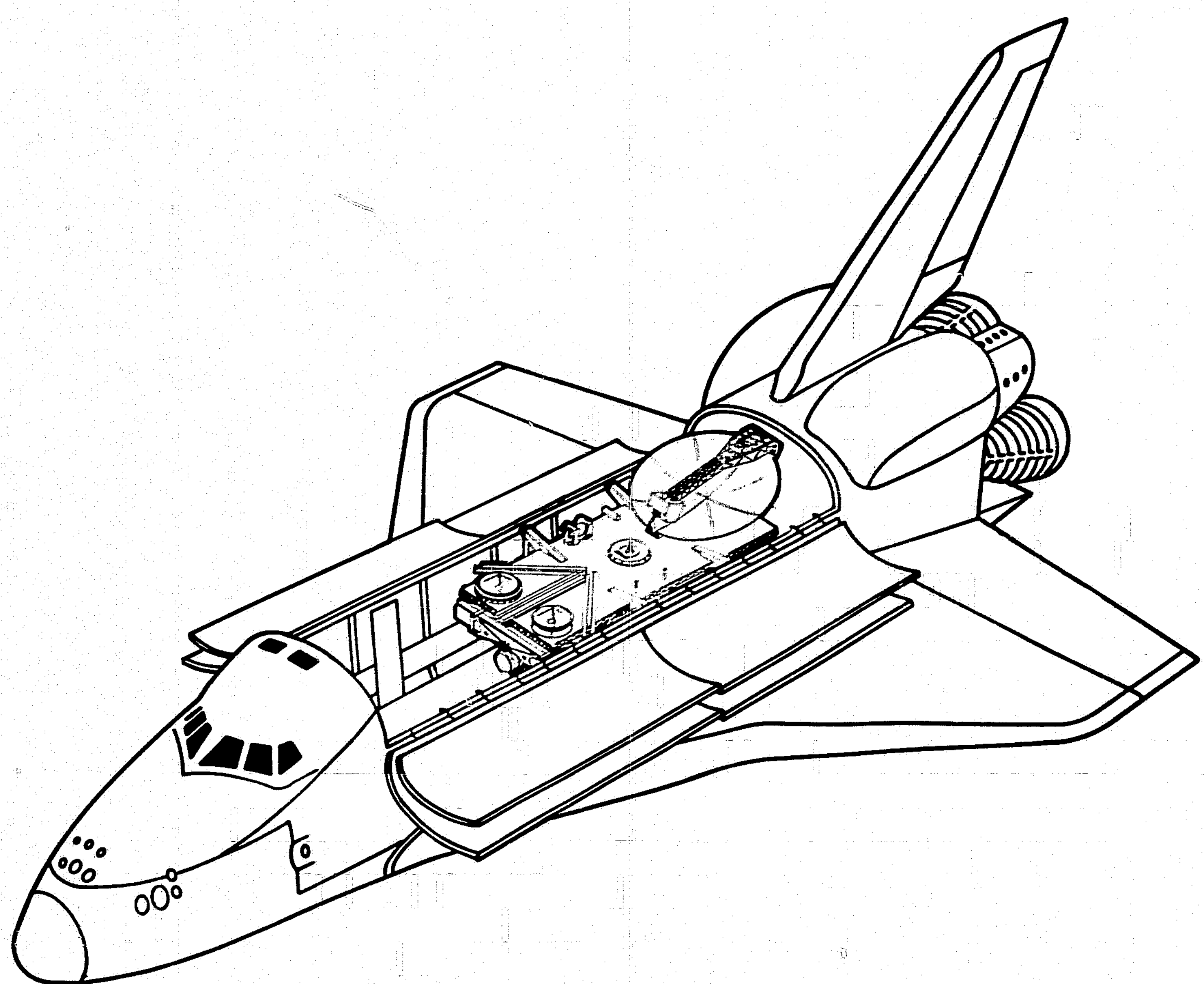


Figure 4.5-11. NOSS Spacecraft Stowed In Orbiter Payload Bay

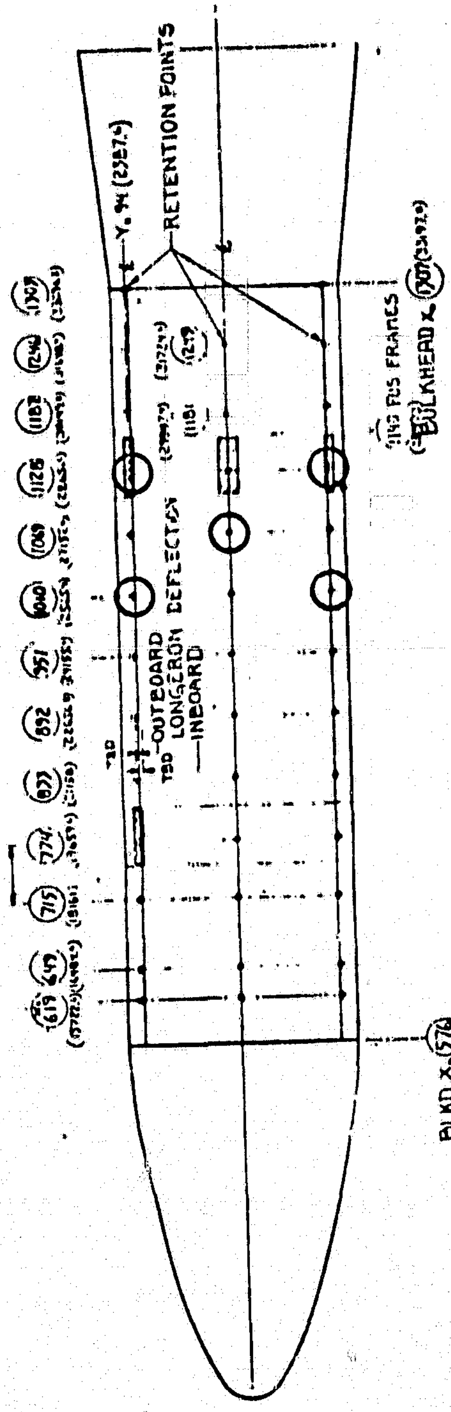


Figure 4.5-12. NOSS Spacecraft Mounting Location In Orbiter

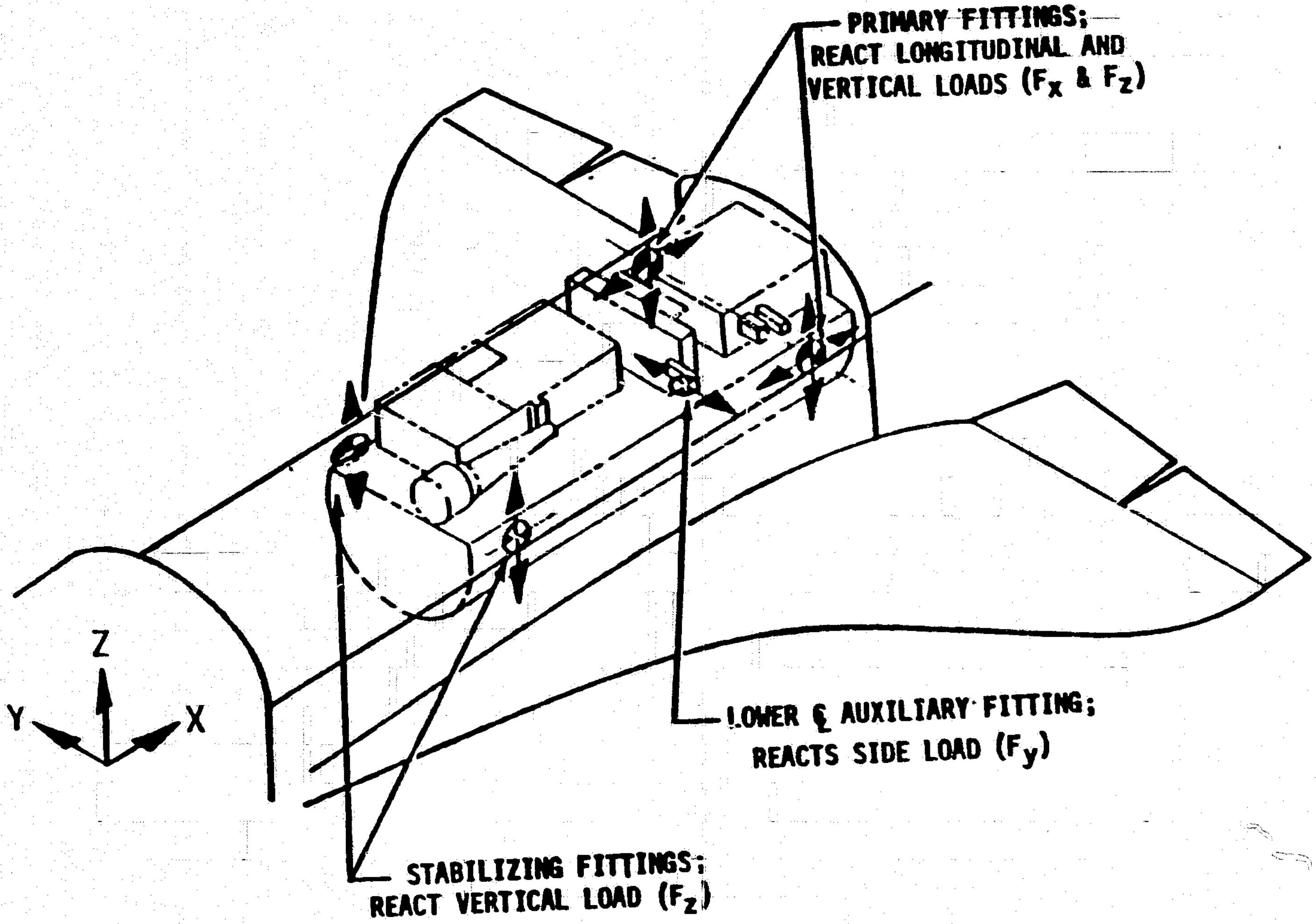


Figure 4.5-13. 5 Point Payload Retention System (Indeterminate)

4.5.4.3 Deployment and Retrieval Requirements

Hardware required for a deployed/retrieved spacecraft includes the orbiter active latch guide system, scuff plates mounted to each trunnion, and the motorized payload umbilical connector discussed in Section 5.0.

The spacecraft is positioned with the LAMMR located aft to provide RMS access and operator visibility to a simply mounted grapple stationed near the center of gravity (X=1062). This grapple location is well within the maximum aft grapple location (X=1085) at which the RMS can lift the maximum size payload from the orbiter bay.

The area around the grapple was left clear to allow sufficient clearance for the RMS grapple. Clearance requirements are specified in Figure 4.5-14 and the NOSS design far exceeds these limits..

4.5.4.4 Orbiter Center of Gravity Restraints

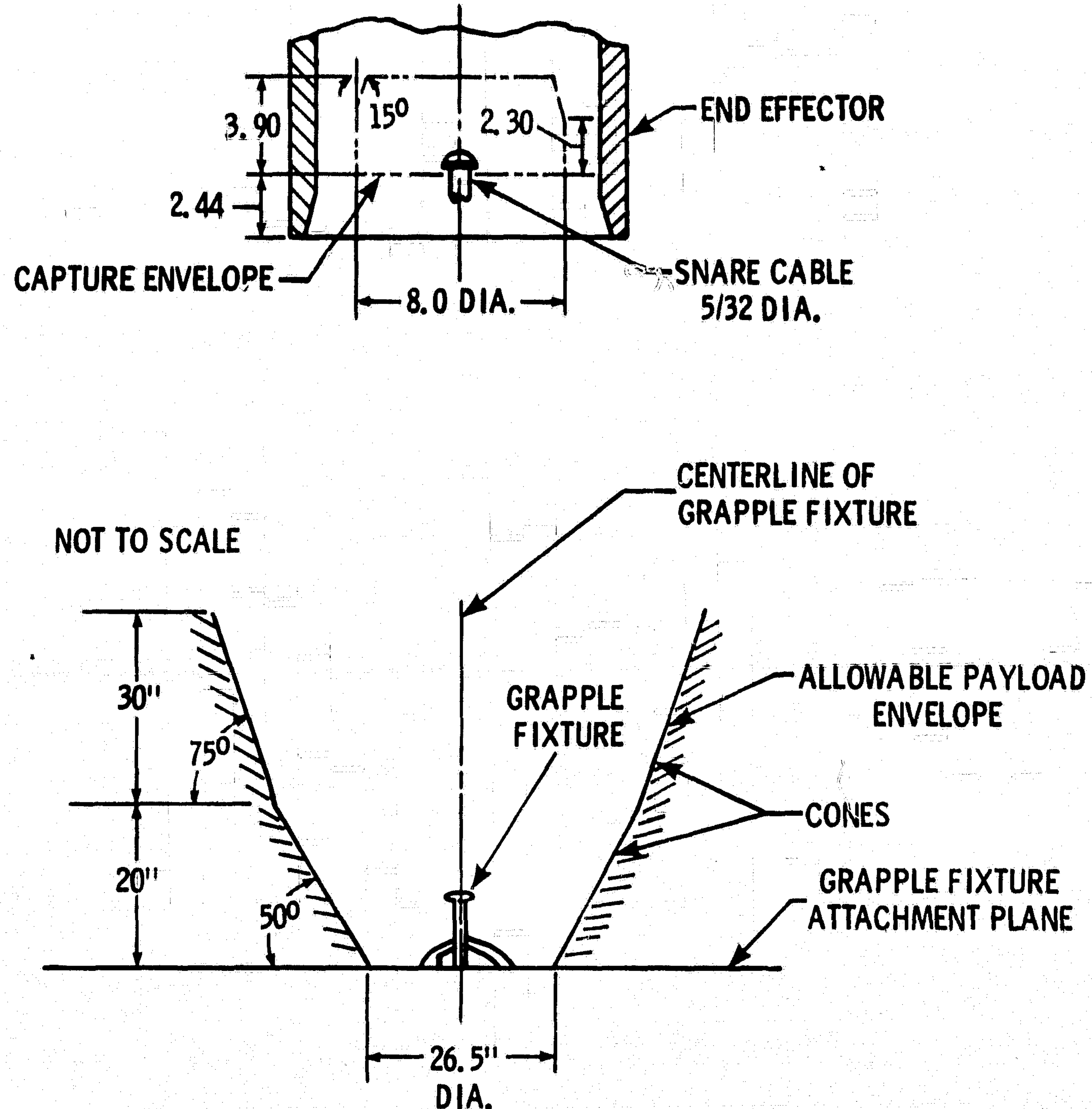
The cargo center of gravity restraints specified by the STS were evaluated for the NOSS spacecraft and are shown for the X-axis, Y-axis and Z-axis in Figures 4.5-15, 4.5-16 and 4.5-17, respectively.

For the launch, retrieval, and abort modes, this analysis showed that the NOSS spacecraft C.G. comfortably meets the cargo limits. The critical limit is the Y-axis and judicious packaging of electronic boxes allows maintenance of an adequate margin.

For retrieval, the propellant expenditure will affect the X and Z C.G. positions; however, these C.G. locations are still within the specified limits.

4.5.5 SPACECRAFT DEPLOYMENTS

The NOSS configuration contains appendages which would violate the shuttle orbiter payload dynamic envelope and/or which are incapable of withstanding the loads associated with launch and retrieval. These elements are folded and latched to the spacecraft or caged until the NOSS is removed from the shuttle

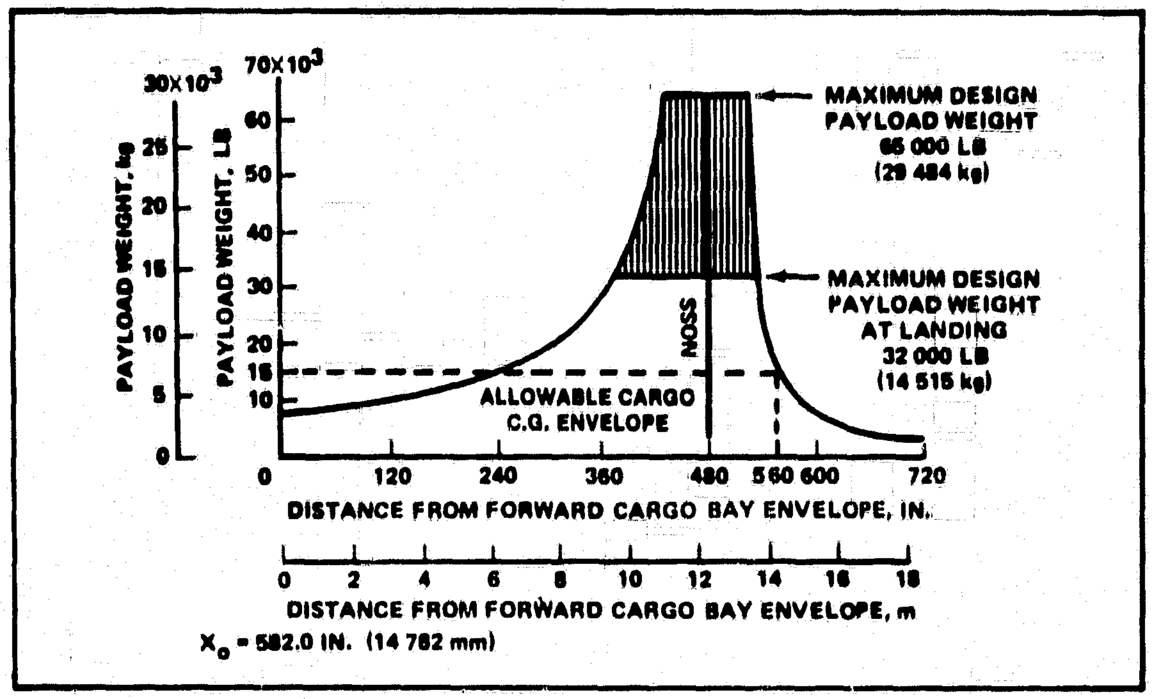


NOTES:

1 CLEARANCE VOLUME CENTERED ON CENTERLINE OF GRAPPLE FIXTURE

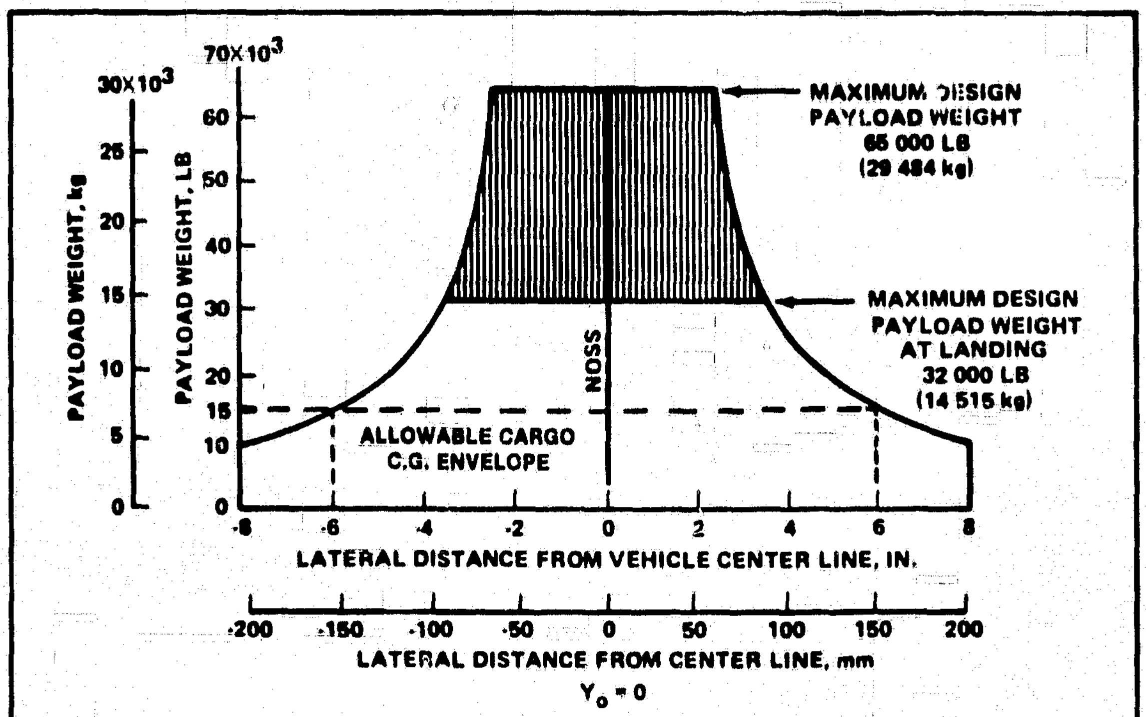
2 CLEARANCES REQUIRED BEYOND 50 INCHES FROM THE ATTACHMENT PLANE WILL BE DEPENDENT ON THE PAYLOAD AND THE REQUIRED ARM CONFIGURATION

Figure 4.5-14. STS/RMS Grapple Clearance Envelope



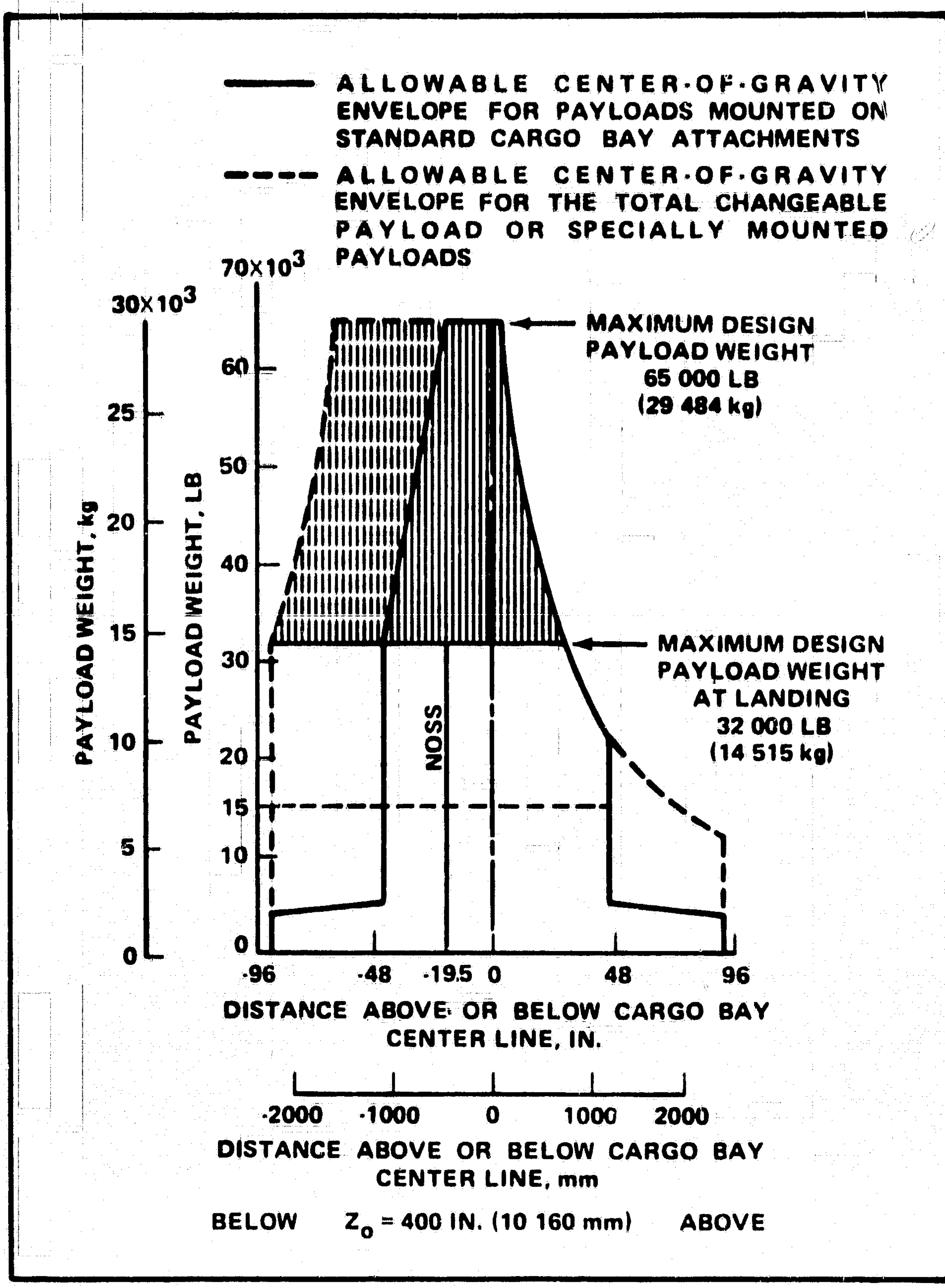
Payload center-of-gravity limits along the X-axis (X_o) of the Orbiter.

Figure 4.5-15. Payload Bay C.G. Location for NOSS - X-Axis



Allowable center-of-gravity envelope along the Orbiter Y-axis (Y_o).

Figure 4.5-16. Payload Bay C.G. Location for NOSS - Y-Axis



Center-of-gravity limits of cargo along the Z-axis (Z_o) of the Orbiter.

Figure 4.5-17. Payload Bay C. G. Location for NOSS - Z-Axis

bay. Deployments occur while NOSS is still attached to the shuttle via the RMS as shown in Figure 4.5-18.

Mechanical devices are used to unlatch, deploy, stow, and relatch equipment that is attached to deployed booms and articulating members. The mechanisms will be actuated on command and during use will not induce excessive loads to the appendage or deployed equipment. The mechanisms control the rate of deployment. These events are accomplished by drive motors and gear trains. Devices are used to indicate the position of each appendage, i.e., deployed or stowed and locked.

NOSS deployments include the LAMMR feed arm, solar array, high-gain antenna, CZCS mirror and radiative cooling door, and CZCS contamination cover.

The primary retrieval mode relies on mechanical devices such as drive motors and gear trains to stow and latch deployed components. A backup capability for retractions is also provided. This backup capability is either EVA or jettison.

4.5.6 FIELDS-OF-VIEW

The NOSS instruments obtain their data through sensing techniques which require several geometric configurations. Communications and data links, tracking and orbit determining systems, and instrument optical and microwave instantaneous fields-of-view must be kept clear of all obstructions over the range of angles through which those fields are scanned or operate. The required clear field is described in terms of an aperature diameter, beam divergence scan geometry, and location of the instrument aperature. Additionally, some fixed fields-of-view must be maintained (e.g., cooling radiators and cold space calibration) for some instruments.

The instruments/antennas require a combination of cross-track and conical scans, calibration and cooling scan/aperature, and fixed beamwidths as shown in Figure 4.5-19. In this figure, the solar array in the horizontal view overlaps the vertical view. Labels such as "solar array horizontal" and "solar array vertical" apply to the vertical and horizontal views in this figure and not to solar array position.

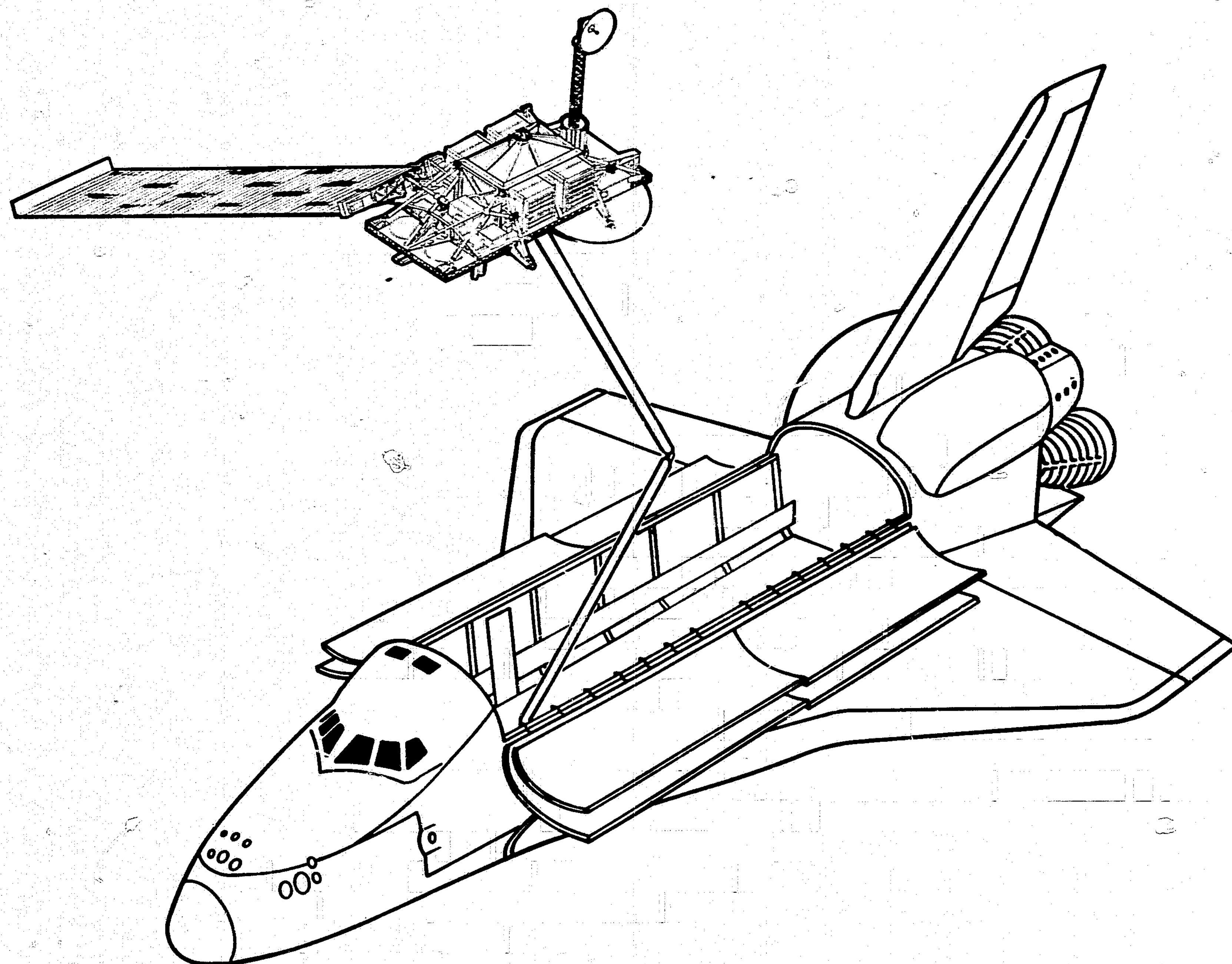


Figure 4.5-18. NOSS Spacecraft Deployed On RMS

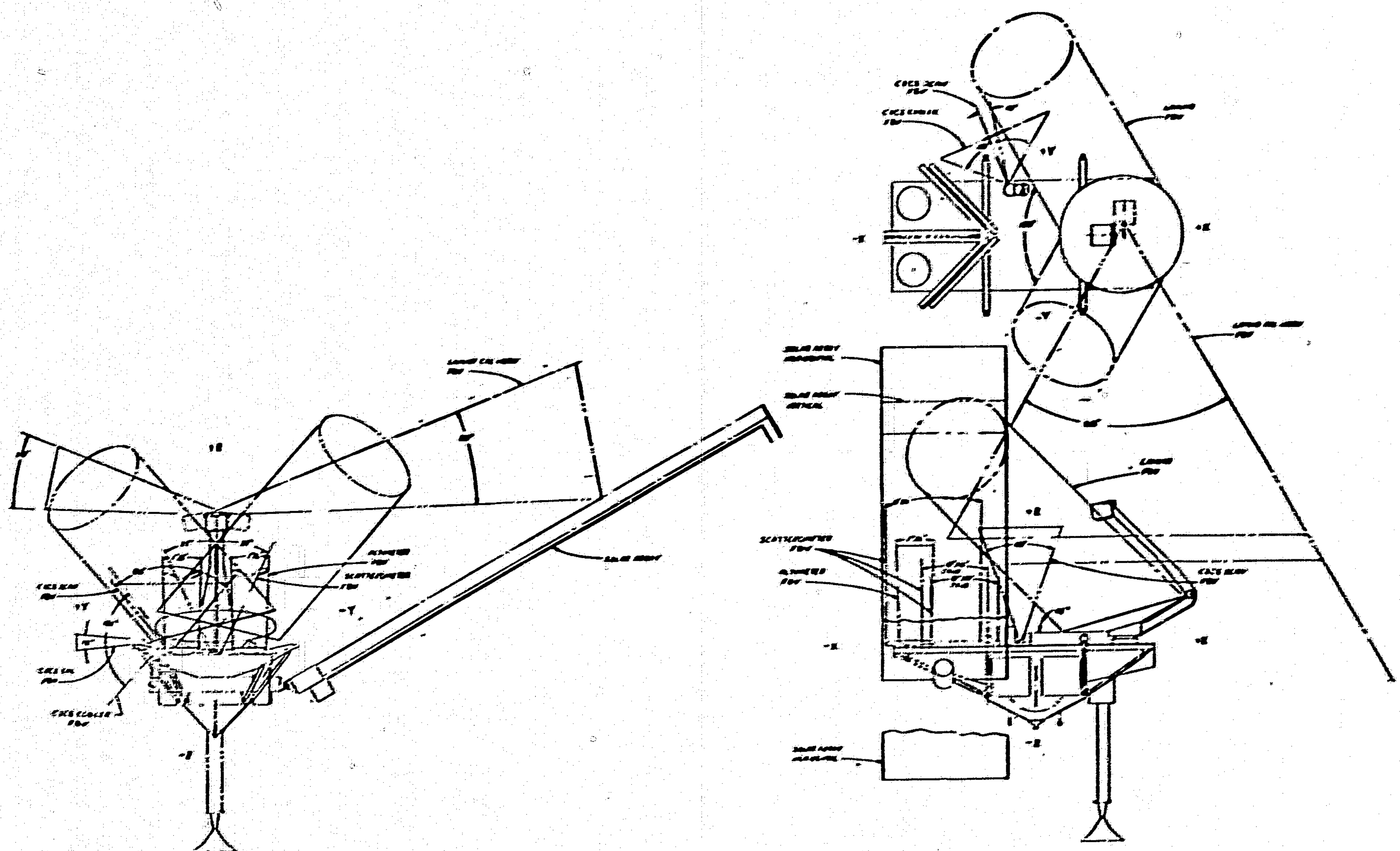


Figure 4.5-19. Instruments Field Of View

The NOSS instruments and antennas for which clear fields-of-view are required include low gain antennas, high-gain antenna, GPS antenna, command and data acquisition antenna, laser retroflector, star trackers, RMS access, LAMMR, CZCS, SCAT, and ALTS.

4.5.6.1 High-Gain Antenna

A 1.8 -m two degree-to-freedom HGA is mounted on an extendable boom and located on the space-side of the spacecraft in a position to view one TDRSS for the majority of the orbit without blockage from any spacecraft appendage. The required field-of-view is a full cone ranging between -10° and the local zenith. The solar array approaches but does not reach the required HGA field-of-view.

4.5.6.2 Low-Gain Antennas

Two low-gain antennas are located on the spacecraft to provide hemispherical coverage with no tracking requirements. The antennas are fixed-mounted, nominally zenith and nadir pointing.

The zenith-oriented LGA is mounted aft and near the shuttle keel fitting and provides nearly unobstructed hemispherical coverage. Obstructions consist of a 20° by 30° forward sector blockage near the local horizon by the keel fitting and blockage due to the solar array. These worst-case blockages are less than 5 per cent.

The nadir-oriented LGA is located near the middle of the instrument platform on the Earth-side of the spacecraft. Obstructions to hemispherical coverage are the LAMMR dish and LAMMR feed in the aft direction; the CZCS in the lateral direction, and the solar array in the forward and lateral directions. The solar array obstructs a 30° by 45° sector (less than 2 percent of the hemisphere) adjacent to the spacecraft horizon, the LAMMR dish obstructs a 30° by 120° sector (3 percent blockage) facing aft, the LAMMR feed and support obstruct a 10° by 30° sector, and the CZCS obstruction is negligible. Total blockage is less than 6 percent.

4.5.6.3 Command and Data Acquisition Antenna

A nadir-oriented omni CDA antenna is located on the instrument platform on the Earth-side of the spacecraft. Obstructions to hemispherical coverage are identical to the nadir-oriented LGA.

4.5.6.4 Global Positioning System Antenna

A zenith-oriented GPS omni antenna is mounted on the space side of the spacecraft forward of the keel fitting. To meet the -10° to local zenith coverage requirement, the antenna must be deployed to extend beyond the keel fitting and solar array. However, an undeployed GPS antenna is used to minimize deployments. Obstructions to the undeployable GPS antenna are the keel fitting in the aft direction and the solar array in the lateral direction. These obstructions are within the 10 per cent obscuration allowable.

4.5.6.5 Laser Retroflector Array

A 60° half-angle full cone unobstructed field-of-view is required for the LRA. A 6° by 60° sector on the aft face of the cone is obstructed for 0.2s by the LAMMR feed during each LAMMR revolution. This blockage is less than one percent of the required field-of-view and will not prevent acceptable performance.

4.5.6.6 Star Tracker

The star tracker has an unobstructed field-of-view.

4.5.6.7 LAMMR

The LAMMR beam is a tapered cylinder originating at the LAMMR dish with a maximum 1.3° beamwidth. Although the antenna scans through a 360° range, the active portion of the scan is during a 120° forward sector ($\pm 60^{\circ}$ of subtrack) of the scan. There are no obstructions.

Instrument calibration field-of-view requirements are a 20° by 60° cold-space sector on each side of the spacecraft originating at the LAMMR calibration horn. The solar array obstructs a 10° by 20° sector on one side but will not prevent acceptable performance.

4.5.6.8 CZCS

The CZCS requires an unobstructed ±40° Earth-viewing and 10° space-viewing sector across track and ±20° along-track field-of-view. There are no obstructions.

The CZCS radiative cooler requires a 101° conical space field-of-view. The forward +Y trunion obstructs a 30° by 30° sector.

4.5.6.9 SCAT

Each of the 6 SCAT antennas has a ½° by 25° fan beam. There are no obstructions.

4.5.6.10 ALT

Each of the altimeters has a tapered-cylindrical beam with a 1.6° beamwidth. There are no obstructions to either beam.

4.5.6.11 Earth Sensor

The Earth sensor requires a 54° half-cone angle. There are no obstructions.

4.6 THERMAL

The thermal design concept is to provide independent temperature control of the structure, each of the major subsystems, and the instrument interfaces. The temperature control techniques utilize multilayer insulation blankets, radiators, redundant heaters with solid-state thermostats, and/or thermal louvers.

4.6.1 ORBITAL ENVIRONMENT

The orbit average unblocked solar, earth albedo and earth infrared fluxes incident on each spacecraft axis are shown in Table 4.6-1 for 705 KM, 11 o'clock, polar, sun synchronous, earth orbit. These fluxes which were generated for the Landsat-D program, are presented as being fairly representative of the environment in the proposed 800 KM, 10:30 orbit. The transient solar and earth albedo fluxes are shown in Figure 4.6-1 and 4.6-2.

Table 4.6-1. Orbit Average Fluxes BTU/HR/FT^2

Axis	Solar	Albedo	Earth IR
+X	95.6	9.1	17.4
-X	95.6	9.1	17.4
+Y	0	8.6	17.4
-Y	72.0	9.6	17.4
+Z	14.1	31.3	59.5
-Z	132.0	0	0

4.6.2 STRUCTURE

The structure is covered with multilayer insulation blankets to minimize temperature gradients within the structure and to maximize the thermal time constant. The blankets are to be constructed with alternating layers of 1/3-mil kapton, aluminized on both sides, and dacron mesh. The outer layers

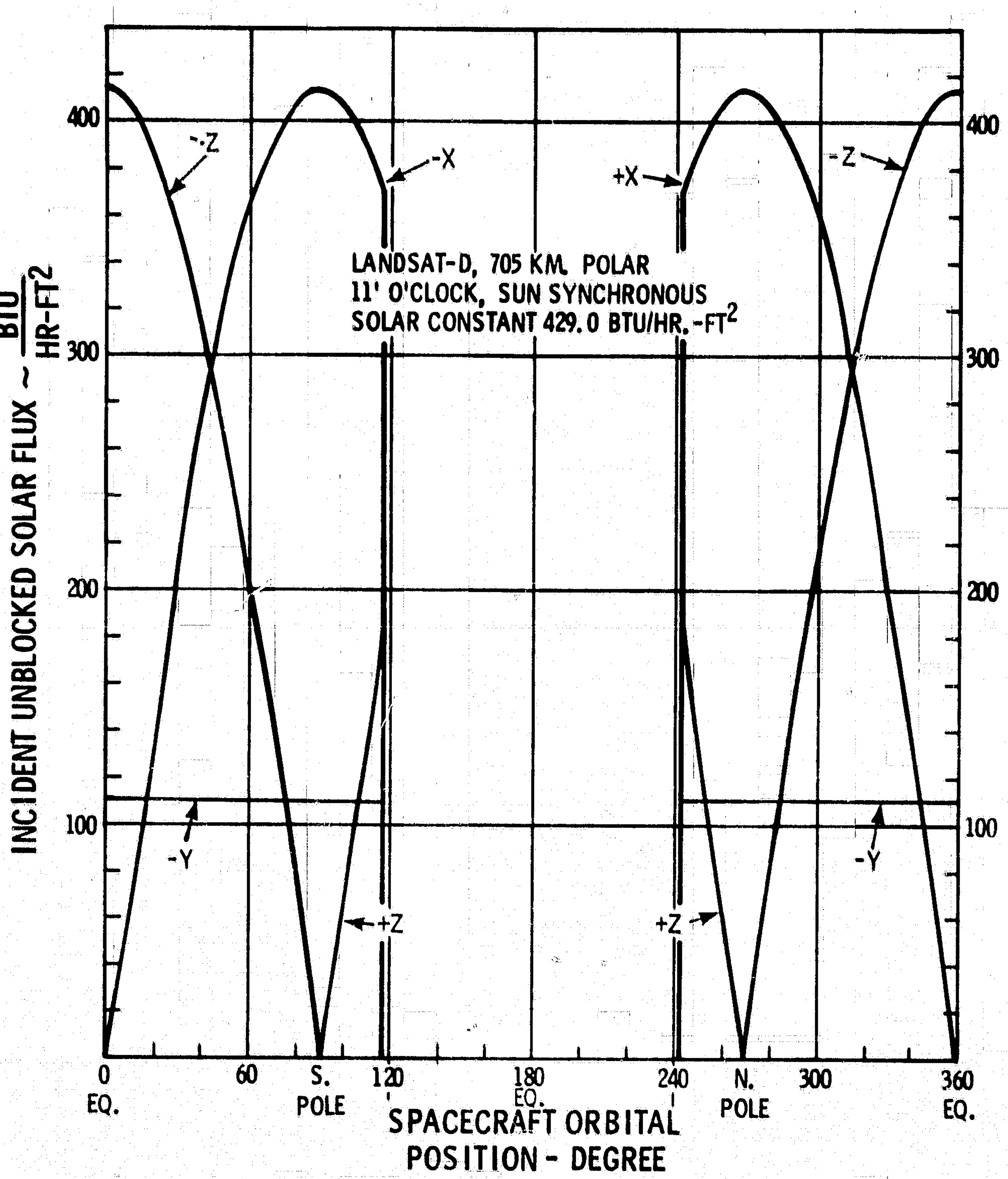


Figure 4.6-1. Incident, Unblocked Solar Flux~$\frac{BTU}{HR.-FT^2}$

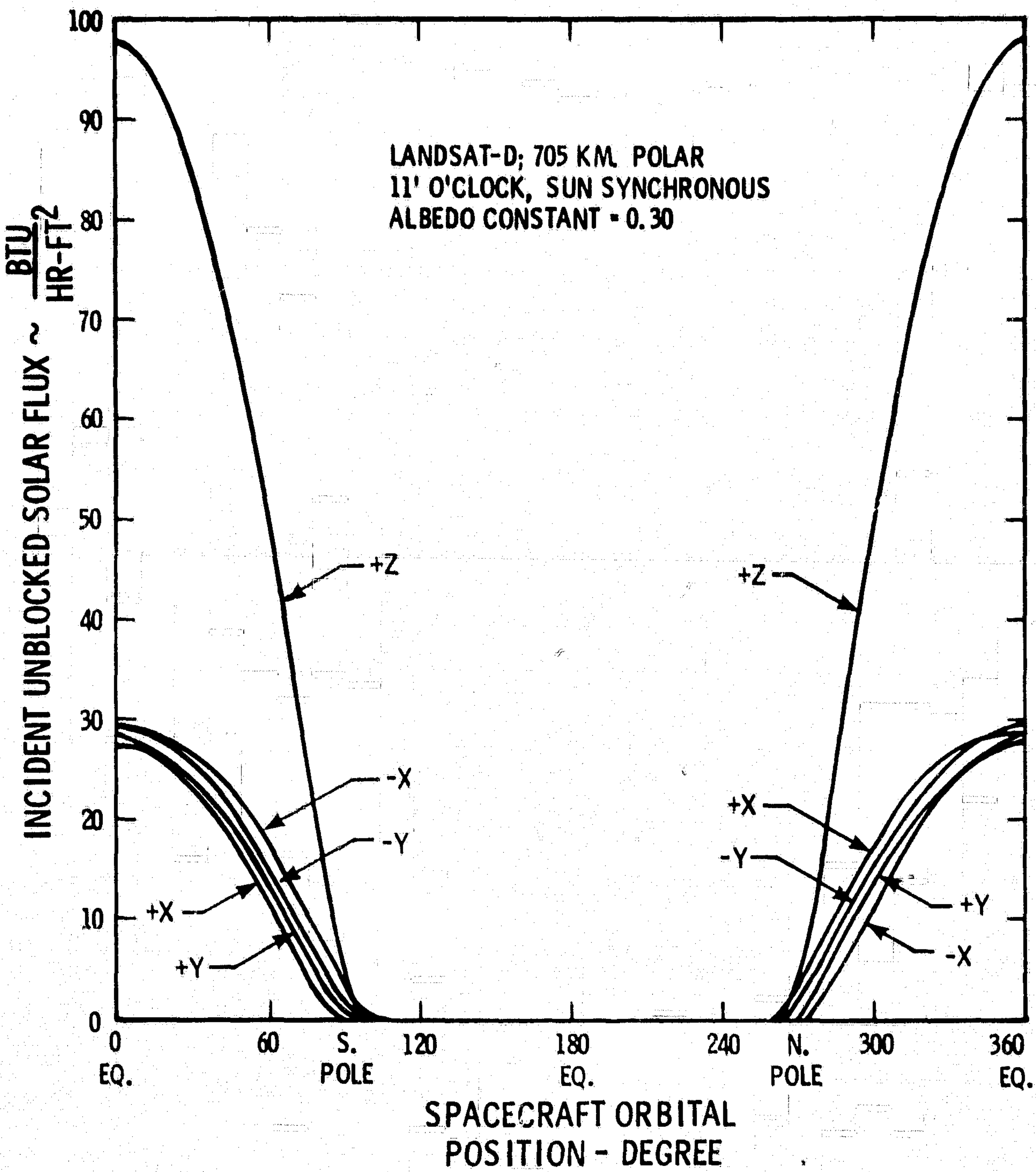


Figure 4.6-2 Incident, Unblocked Albedo Flux ~ $\frac{BTU}{Hr.-Ft.^2}$

are to be 3-mil kapton aluminized on one side, with the kapton side out. If it becomes necessary to provide diffusely reflecting external surfaces for the shuttle, beta cloth can be attached to or substituted for the 3-mil kapton outer layer. Blanket weight is estimated to be 240 lbs, based on 0.2#/ft^2 with a beta cloth outer layer, 900 ft^2 of surface area, and 60 lbs for attachment hardware.

The top and bottom surfaces of the platform make up the bulk of the exposed surface area. With a 3-mil kapton outer surface ($\alpha = .47$, $\epsilon = .87$), the average structure temperature would be approximately -15°C without any source of heat except for the orbital environment. With a blanket effective emittance of 0.02 $\pm$ .01, the heater power required to regulate the structure temperature +10°C is estimated to 180 $\pm$ 90 watts. The thermal time constant is estimated to be 15 days.

4.6.3 PROPULSION SYSTEM

The hydrazine tank and the lines are thermally isolated from the structure and wrapped with multilayer insulation. Tank heater requirements are estimated to be 35 $\pm$ 17.5 watts to control the temperature to +10°C, assuming that the tank surface area is 130 ft^2, the insulation effective emittance is 0.02 $\pm$ .01, and the average outer layer temperature is -25°C. Heat conduction between the tank and the structure is assumed to be negligible because of the thermal isolation mounts and minimal temperature gradients. With 3800 lbs of hydrazine in the tank, the thermal time constant is estimated to be 58 days.

The heater requirements for the hydrazine lines are estimated to be 0.13 watts/ft of line. For 100 ft of lines, the total line heater power is 13 watts.

The thrusters are enclosed in 8 modules, each with independent temperature control. Heater power requirements are estimated to be 10 watts per module.

4.6.4 MODULAR POWER SYSTEM

The two MPS modules are mounted with the louvers viewing the +Y direction to minimize the environmental heat loads. Each module has two radiators, each with a thermal louver assembly. The upper radiator and louver assembly regulate the temperatures of the power regulator unit, the power control unit and the signal conditioner assembly. The lower radiator and louver regulates the temperatures of the batteries. The maximum orbit average heat dissipation for each MPS module is estimated to be 163 watts in the upper radiator and 96 watts in the batteries. The thermal environment of the MPS modules for the NOSS mission is about 60% less severe than the hot case environment specified for the MPS design. Thus, there is some margin for increased power dissipation.

4.6.5 ACS MODULE

The primary thermal control surface views the -Y direction. To avoid exposure to sunlight, the louvers are covered with a silver-teflon coated panel. Since for NOSS the two star trackers, which are normally mounted in the ACS module, are to be housed separately near the LAMMR, the ACS module orbit average power is estimated to be about 95 watts. This power level is well within the capabilities of the ACS thermal design for the NOSS environmental heat loads.

4.6.6 C&DH MODULE

This module also views the -Y direction. The estimated power dissipation of 100 watts is also well within the capabilities of the thermal control system for the NOSS environmental heat loads.

4.6.7 INSTRUMENT INTERFACES

Interface requirements for most instruments have not been defined at this time to the detail required for the thermal design of the interfaces.

However, it is possible to discuss a design concept which can accommodate an instrument whether it is designed specifically for NOSS or for another spacecraft. The concept is to provide thermal isolation between the instrument mount and the NOSS structure, to control the interface temperature with heaters when the instrument is turned off, and to provide a radiator sized to handle the heat transfer from the instrument to the interface when the instrument is operating. In general, this concept is to be applied to each instrument, module or box which is attached to the structure.

Detailed thermal modeling studies are required to optimize the interface design and to size the radiators and heaters. Design trade-offs are possible which can simplify the thermal designs on either side of an interface and minimize the heater power requirements. The major trade-off is to relax the degree of thermal isolation to the extent that heat flow across the interface does not interfere with the temperature control of the structure or cause excessive temperature gradients and thermal distortions.

4.6.8 SHUTTLE THERMAL ENVIRONMENT

Detailed thermal modeling of temperature transients is required for NOSS in the shuttle bay during launch and landing, in orbit with the payload bay doors open, during deployment and recovery when attached to the RMS, and during transfer to and from the 800 KM orbit. For the sequence of events as described in Section 5, Mission Timeline, the temperature changes of the structure and the hydrazine tanks are expected to be negligible because of their relatively long thermal time constants. For the electronics in the experiments, modules and assorted boxes the thermal responses to power turn on or off or to significant changes in environment heat loads are expected to be in the order of a few degrees per hour. Some constraints on spacecraft or shuttle attitude may be required to avoid prolonged exposure to unfavorable environments. Potential problems anticipated at this time and tentative solutions being considered are (1) the effects of sunlight in the shuttle bay, which can be prevented by curtains or shields attached either to NOSS or to the shuttle, and (2) shuttle bay contamination of optical surfaces, which can be prevented by removable covers.

5.0 MISSION TIMELINE

A mission timeline for the NOSS program was generated for the pre-launch, launch, shuttle orbit, orbit transfer, and shuttle retrieval activities. The assumptions and guidelines used to derive this timeline included:

a) Shuttle Launch at WTR (VAFB).

b) Spacecraft shipped intact to WTR.

c) Minimum spacecraft checkout activities at WTR.

d) Minimize series activities with STS operations at WTR.

e) Perform preliminary spacecraft checkout after removal from the orbiter.

f) Deploy spacecraft system (solar array, HGAS, LAMMR arm) while on RMS and complete spacecraft checkout.

g) Orbiter remains in parking orbit until spacecraft checkout and orbit transfer is confirmed.

h) Orbit transfer accomplished with continuous burn of large thrusters.

i) Spacecraft is retrieved.

Time was estimated from proposed STS timelines and operational flows.

5.1 PRE-LAUNCH OPERATIONS

The processing flow for automated spacecraft is shown in Figure 5.1-1 (Reference "NASA/VAFB STS Payload Cargoes Ground Operation Plan", K-CM-09.1, July 1979, Review copy) and a timeline presented in Table 5.1-1. The facilities and ground operations plan for WTR are still in the planning stages. The NOSS pre-launch processing flow differs from the current baseline for WTR in that the

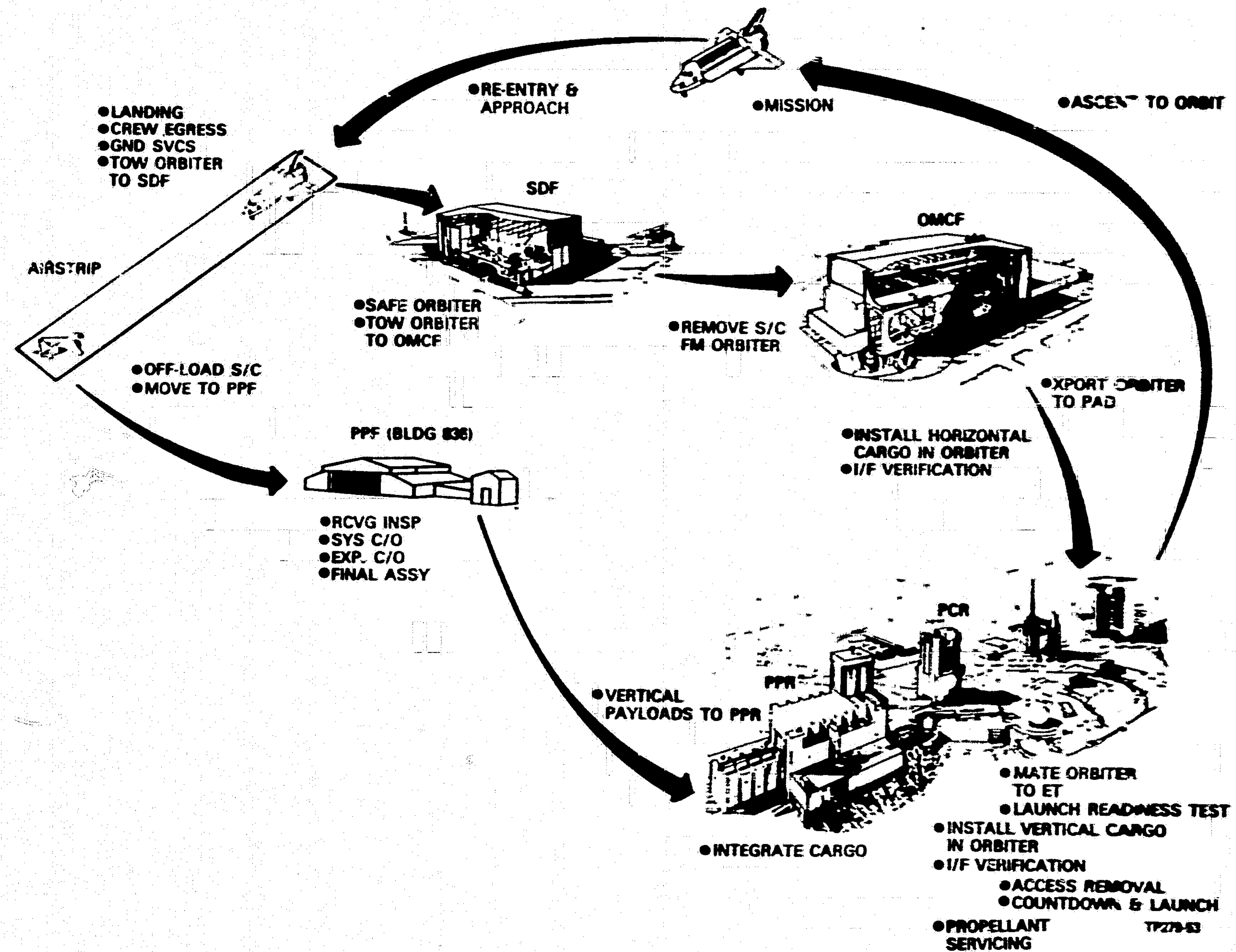


Figure 5.1-1. Automated Spacecraft Baseline VAFB Processing Flow

propellant loading for the NOSS spacecraft is scheduled for the Payload Preparation Room (PPR) rather than the Hazardous Processing Facility (HPF). Interface verification is accomplished with the CITE in the PPR.

Prior to launch, the spacecraft is configured for flight, through the T-0 umbilical, by turning on the battery charging section of the MPS, the transponders and OBC in the C&DH module, and the ACS is maintained in a standby mode. Monitoring of the propulsion subsystem will be accomplished by STS via the spacecraft umbilical. This umbilical is mounted on the side of the STS and has an in-flight disconnect connector. The umbilical is motor driven for retraction and re-insertion during retrieval.

TABLE 5.1-1

NOSS MISSION TIMELINE

PRE-LAUNCH OPERATIONS TIMELINE (VANDENBURG AIR FORCE BASE)	TIME OF INITIATION DAY:HOUR:MINUTE	DURATION	RESPONSIBLE AGENCY
1. SITE ACTIVATION	T-41:45:40	4 days*	NOSS
2. S/C TEST EQUIPMENT RECEIPT AND VALIDATION	T-37:54:40	3 days	NOSS
3. GSE RECEIPT AND SET-UP	T-37:45:40		NOSS
4. S/C ARRIVE AT PPF	T-34:45:40	--	NOSS
5. OFF LOAD S/C AND SET-UP IN PPF	T-34:45:40	1 day	NOSS
6. S/C REC INSPECTION, ALIGNMENT CHECKS AND CABLE HOOK-UP	T-33:45:40	7 days	NOSS
7. S/C PERFORMANCE TESTS	T-26:45:40	10 days	NOSS
8. MOVE TO PPR	T-16:45:40	4 hours	JOINT
9. S/C INTERFACE VERIFICATION (CITE)	T-16:48:40	2 days	JOINT
10. LOAD HYDRAZINE	T-14:45:40	5 days	NOSS
11. WEIGHT	T-09:45:40	1 day	NOSS
12. COMPLETE P/L C/O AND PREP	T-08:41:40	4 days	JOINT
13. TRANSFER TO PCR	T-04:41:40	4 hours	JOINT
14. ROTATE TO VERTICAL	T-04:37:40	2 hours	JOINT

15.	FINAL CARGO PRFPARATION	T-04:35:40	4 days	JOINT
16.	MOVE PCR TO ORBITER (VERTICAL INTEGRATION)	T-00:35:40	4 hours	JOINT
17.	INSTALL S/C IN ORBITER	T-00:31:40	27.5 hours	JOINT
18.	S/C IN-LAUNCH CONFIGURATION	T-00:04:10	4 hours	JOINT

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* DAY = 8 HOUR FOR P/L PREP
16 HOUR FOR STS INTEGRATION

5.2 LAUNCH OPERATIONS

The launch operations including placing the shuttle into its orbit and preparing the spacecraft for deployment are presented in Table 5.2-1. This timeline was developed assuming that the spacecraft is lifted from the shuttle bay by the RMS. A schematic of the launch/orbit sequence is shown in Figure 5.2-1 and the spacecraft is shown in the shuttle bay just prior to lift-out in Figure 5.2-2. Preparation of the spacecraft for deployment is accomplished in parallel with the orbiter state vector up-date.

TABLE 5.2-1

NOSS MISSION TIMELINE

	LAUNCH/FLIGHT OPERATIONS	TIME OF INITIATION DAY:HOUR:MINUTE	DURATION	RESPONSIBLE AGENCY
19.	PRE-LAUNCH POWER CHANGEOVER (GROUND TO STS) AND TURN-ON GYROS	T-00:00:10		STS
20.	LIFT-OFF	00:00:00		STS
21.	MECO	00:00:08		STS
22.	COMPLETE STS ORBIT INSERTION	00:00:45		STS
23.	VERIFY SAFE ORBIT	00:00:45		STS
24.	RECONFIGURE ORBITER GPC	00:00:50	12 min.	STS

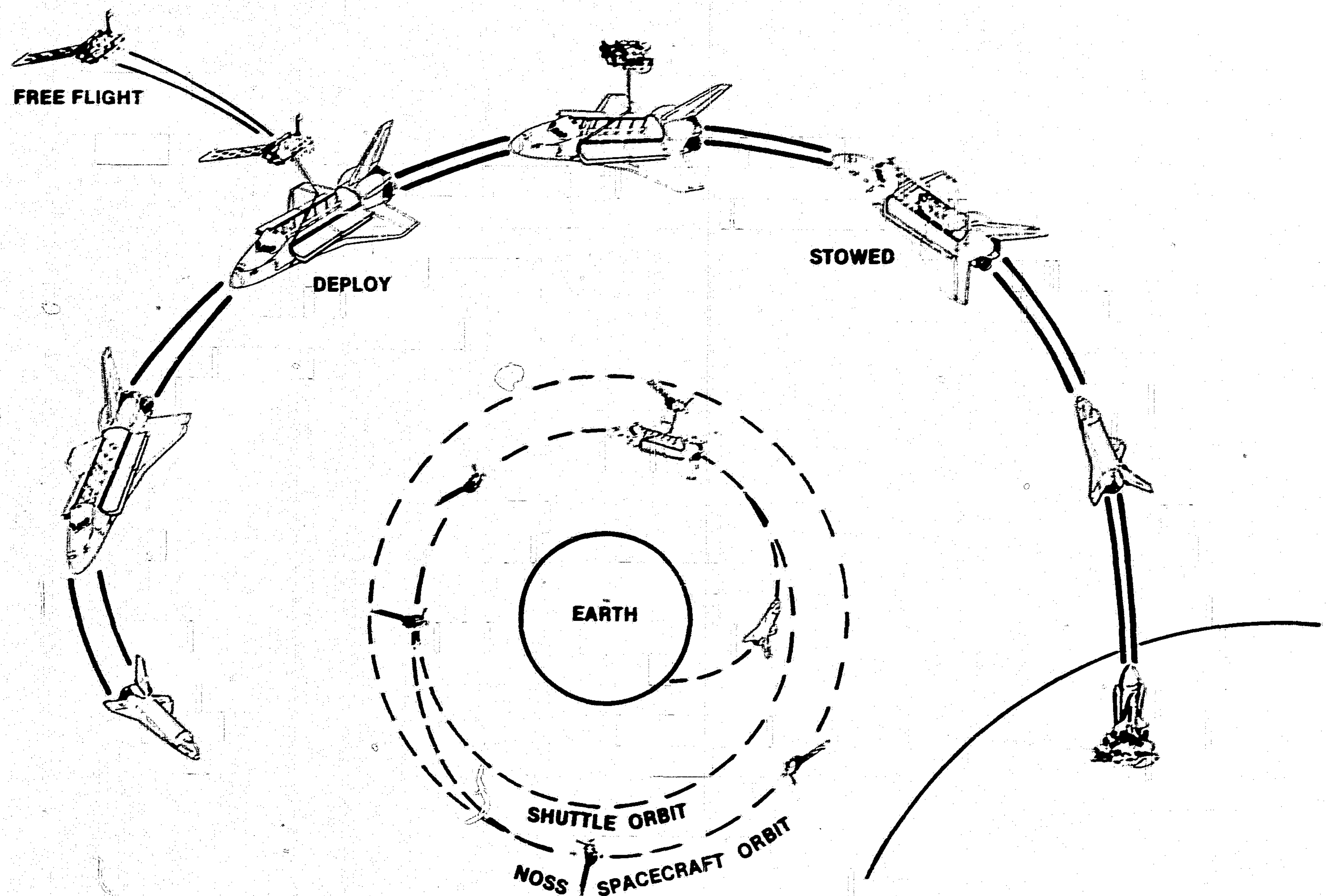


Figure 5.2-1. NOSS LAUNCH/ORBIT SEQUENCE

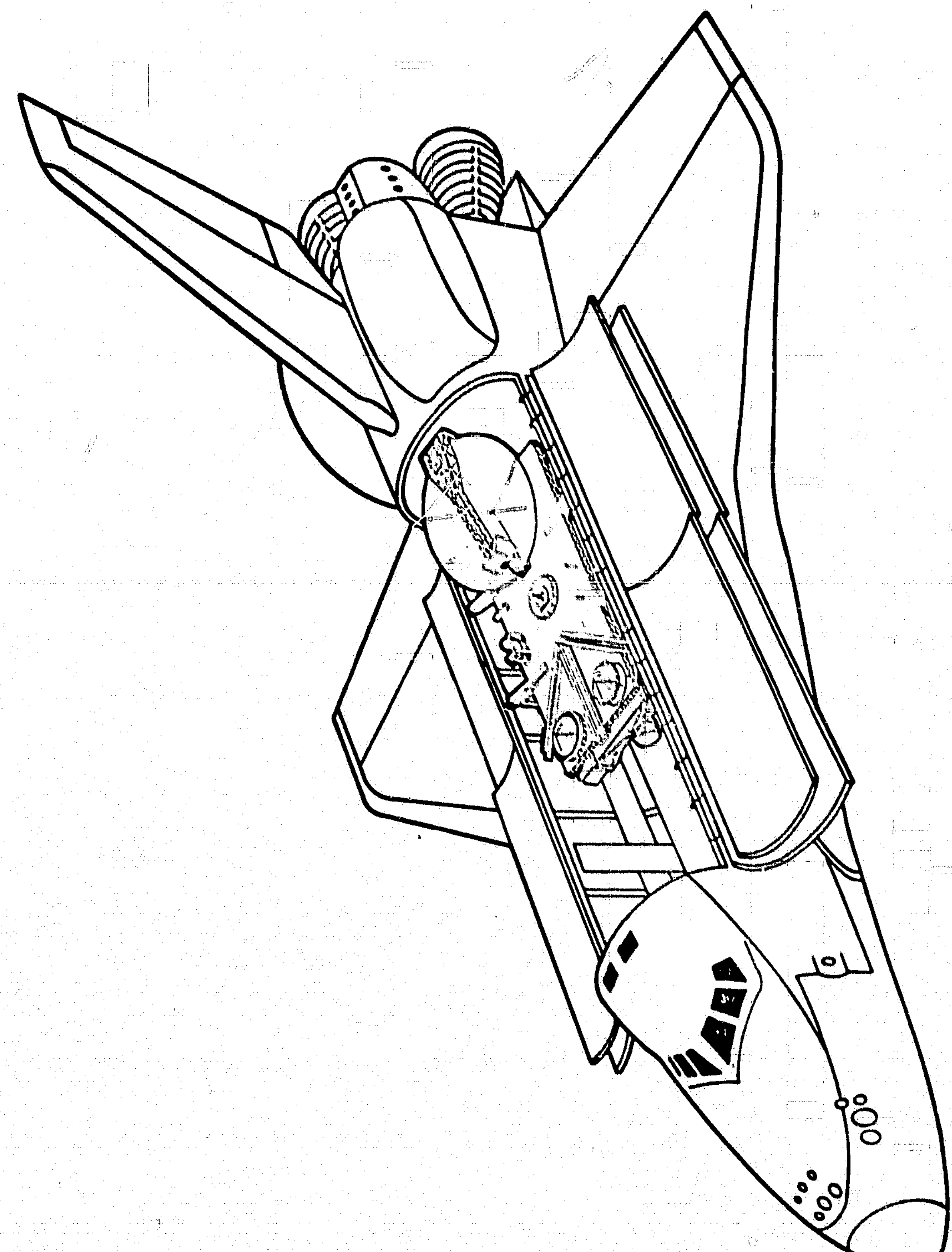


Figure 5.2-2. NOSS Spacecraft Stowed In Orbiter Payload Bay

25.	OPEN PAYLOAD BAY DOORS	00:01:02	4.8 min.	STS
26.	ORBITER COOLING ACTIVATION	00:01:06.8	2.3 min.	STS
27.	GENERATE ORBITER STATE VECTOR UP-DATE DATA	00:01:06.8	(2.5 hrs.)	STS
28.	ACTIVATE AND CHECKOUT KU-BAND COMMUNICATION LINK	00:01:06.8	(3.0 min.)	STS

<u>PERFORM PREDEPLOYMENT FUNCTIONS</u>

29.	ACTIVATE RMS AND CONNECT TO S/C	00:01:10	10 min.	STS
30.	VERIFY PROP SYSTEM: LATCH VALVES POSITION; THRUSTER VALVES STATE	00:01:20	2 min.	NOSS
31.	DISCONNECT STS POWER AND T/M UMBILICAL-WITHDRAW AND STOW UMBILICAL	00:01:22	10 min.	JOINT
32.	DISCONNECT ATTACH FITTINGS	00:01:32	1 min.	STS

5.3 DEPLOYMENT OF SPACECRAFT

The RMS lifts the spacecraft out of the shuttle bay and turns it over so that the top or space viewing side of the spacecraft is pointed away from the orbiter. The communication link with TDRSS is activated and the health of the spacecraft is confirmed and an initial stellar attitude up-date is performed.

Deployment of the spacecraft equipment and instruments is initiated. The solar array is deployed, the power module activated and checked out. The Hi-Gain Antenna, the LAMMR arm and the CZCS and CZCS cover are deployed. The spacecraft checkout is then completed and the decision to release the spacecraft is made.

Since the RMS has little or no capability to impart a ΔV to the spacecraft at separation, the baseline specifies a spacecraft maneuver to achieve the desired separation velocity and distance between the spacecraft and orbiter. The ACS and NPS is activated and the small thrusters (.5 lb) are used to maintain spacecraft attitude and provide the separation velocity between the spacecraft and the orbiter.

After the two vehicles have reached a separation distance of approximately 10 NM, the main thrusters can be activated for orbit transfer. A time of 45 minutes has been set to accomplish this separation.

The shuttle remains in parking orbit until spacecraft orbit transfer has been accomplished. The timeline for this operational phase is shown in Table 5.3-1.

TABLE 5.3-1

NOSS MISSION TIMELINE

PERFORM DEPLOYMENT	TIME OF INITIATION DAY:HOUR:MINUTE	DURATION	RESPONSIBLE AGENCY
33. LIFT S/C FROM P/L BAY WITH RMS	00:01:33	10 min.	STS
34. ACTIVATE RF TRANSMITTER AND C/O C&DH AND RF S/S. ESTABLISH TDRSS LINK	00:01:43	20 min.	NOSS
35. PREL C/O S/C AND PERFORM S/C ATTITUDE UP-DATE	00:02:03	30 min.	NOSS
DEPLOY AND ACTIVATE S/C SUBSYSTEMS			
36. SOLAR ARRAY - DEPLOY	00:02:33	2 min.	NOSS
37. HGAS - DEPLOY ANTENNA	00:02:35	2 min.	NOSS
38. LAMMR - DEPLOY ARM	00:03:37	1 min.	NOSS
39. CZCS - UNCAGE MIRROR	00:03:36	1 min.	NOSS
40. COMPLETE S/C C/O AND ANALYZE DATA	00:03:37	1 hr.	NOSS
41. COMMIT TO RELEASE	00:04:37	2 min.	NOSS
42. RELEASE	00:04:39		STS
43. RETRACT AND STOW RMS	00:04:39	10 min.	STS
44. ACTIVATE S/C ACS AND NPS AND PERFORM SEPARATION MANEUVER	00:04:49	45 min.	STS

45.	VERIFY SAFE SEPARATION DISTANCE	00:05:01	JOINT
46.	SHUTTLE REMAINS IN PARKING ORBIT		STS

5.4 ORBIT TRANSFER

Orbit transfer requirements are discussed in detail in Section 4.2 and 4.3 and the sequence of events are summarized in Table 5.4-1. The orientation of the spacecraft for the orbital mode and both orbit transfer modes, initial and retrieval, are schematically shown in Figure 5.4-1.

TABLE 5.4-1

NOSS MISSION TIMELINE

ORBIT TRANSFER		TIME OF INITIATION DATE:HOUR: MINUTE	DURATION	RESPONSIBLE AGENCY
47.	ACTIVATE S/C PROP S/S (RCS AND MAIN ENG) PULSE/CHECKOUT JETS	00:05:34	5 min.	NOSS
48.	ROTATE S/C 90° TO ALIGN VELOCITY VECTORS	00:05:39	1.2 min.	NOSS
49.	PERFORM S/C STELLAR ATTITUDE UPDATE	00:05:40	5 min.	NOSS
50.	COMMAND ACTIVATION OF PROPULSION FOR ORBIT TRANSFER BURN	00:05:45	60 min.	NOSS
51.	VERIFY MISSION ORBIT	00:06:45	40 min.	NOSS
52.	ROTATE S/C 90° FOR CORRECT ORBIT ATTITUDE	00:07:25	1.2 min.	NOSS
53.	RELEASE SHUTTLE TO RETURN FOR LANDING	00:07:26		NOSS/STS
54.	REMOVE INJECTION ERRORS AND ORBIT ATTITUDE FINE ADJUSTMENT	00:07:26	120 min.	NOSS
55.	ACTIVATE INSTRUMENTS AND C/O	00:09:26	1 wk.	NOSS
56.	INITIATE NORMAL ORBIT OPERATIONS	07:09:26		NOSS

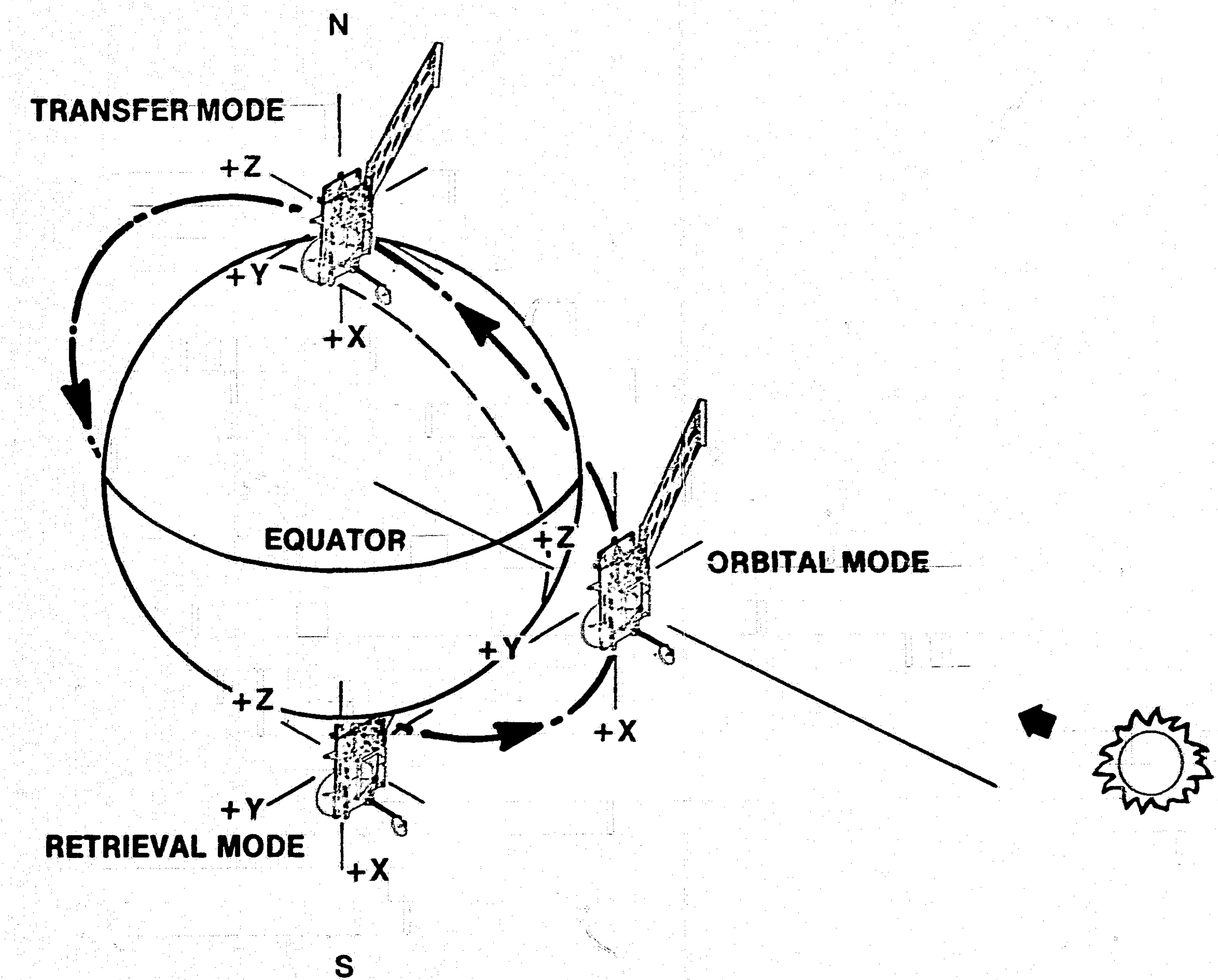


Figure 5.4-1 Spacecraft Orientation For Various Operational Modes

5.5 SPACECRAFT RETRIEVAL

A spacecraft retrieval sequence is presented in Table 5.5-1 and graphically shown in Figure 5.5-1. It was assumed that the spacecraft would remain at its operational altitude until a firm orbiter launch date was obtained. T days prior to the shuttle launch, the spacecraft is rotated to the proper attitude for transfer from its operational orbit to the anticipated shuttle orbit (no inclination angle change is assumed). Orbit transfer is effected and the spacecraft is reoriented and placed in a limited operational mode to await shuttle rendevous. The shuttle initiates final approach and maintains station-keeping to assure that the spacecraft is within RMS capture constraints. The spacecraft is captured and the appendages are stowed while on the RMS. This aids the back-up EVA mode for stowage/ejection of appendages which fail to retract/stow properly. The spacecraft is then positioned in the payload bay, the spacecraft STS umbilical is hooked up and the berthing and stowing of the spacecraft and RMS is completed. The Shuttle Bay doors are then closed and the re-entry sequence is initiated.

TABLE 5.5-1

NOSS MISSION TIMELINE

SPACECRAFT RETRIEVAL	TIME OF INITIATION DAY:HOUR:MINUTE	DURATION	RESPONSIBLE AGENCY
57. UPON COMPLETION OF MISSION, PREPARE S/C FOR DEORBIT, ROTATE S/C 90°	00:00:00	90 min.	NOSS
58. ACTIVATE PROPULSION SYSTEM FOR ORBIT TRANSFER BURN	00:01:30	90 min.	NOSS
59. POSITION S/C AT ORBIT ALTITUDE AND INCLINATION FOR STS RETRIEVAL, ROTATE S/C, AWAIT RETRIEVAL	00:03:00	120 min.	NOSS
60. PREPARE S/C FOR RETRIEVAL - PLACE S/C IN LIMITED POWER MODE - ACS S/S ON	T+00:05:00	30 min.	NOSS

60.	PREPARE S/C FOR RETRIEVAL PLACE S/C IN LIMITED POWER MODE - ACS S/S ON	T+00:05:00	30 min.	NOSS
61.	STS INITIATES FINAL APPROACH - MAINTAINS STATIONKEEPING PERIOD TO ENSURE S/C WITHIN RMS CAPTURE CONSTRAINTS	T+00:05:30	180 min.	STS
62.	RMS OPERATOR ACQUIRES AND TRACKS S/C END EFFECTOR BROUGHT TOWARD GRAPPLE FIXTURE, GRAPPLE TARGET, USE TO ALIGN END EFFECTOR, CAPTURE COMMAND GIVEN WHEN GRAPPLER IS WITHIN CAPTURE ENVELOPE - ACS DEACTIVATED	T+00:08:30	60 min.	STS/NOSS
63.	STOW/EJECT DEPLOYABLE APPENDAGES	T+00:09:30	30 min.	JOINT
64.	POSITION S/C IN SHUTTLE BAY, HOOK-UP UMBILICAL, BERTH AND STOW, (TURN OFF S/C)	T+00:10:06	60 min.	STS
65.	SHUTTLE BAY DOORS CLOSE - RE-ENTRY SEQUENCE INITIATED.	T+00:10:60	5 min.	STS

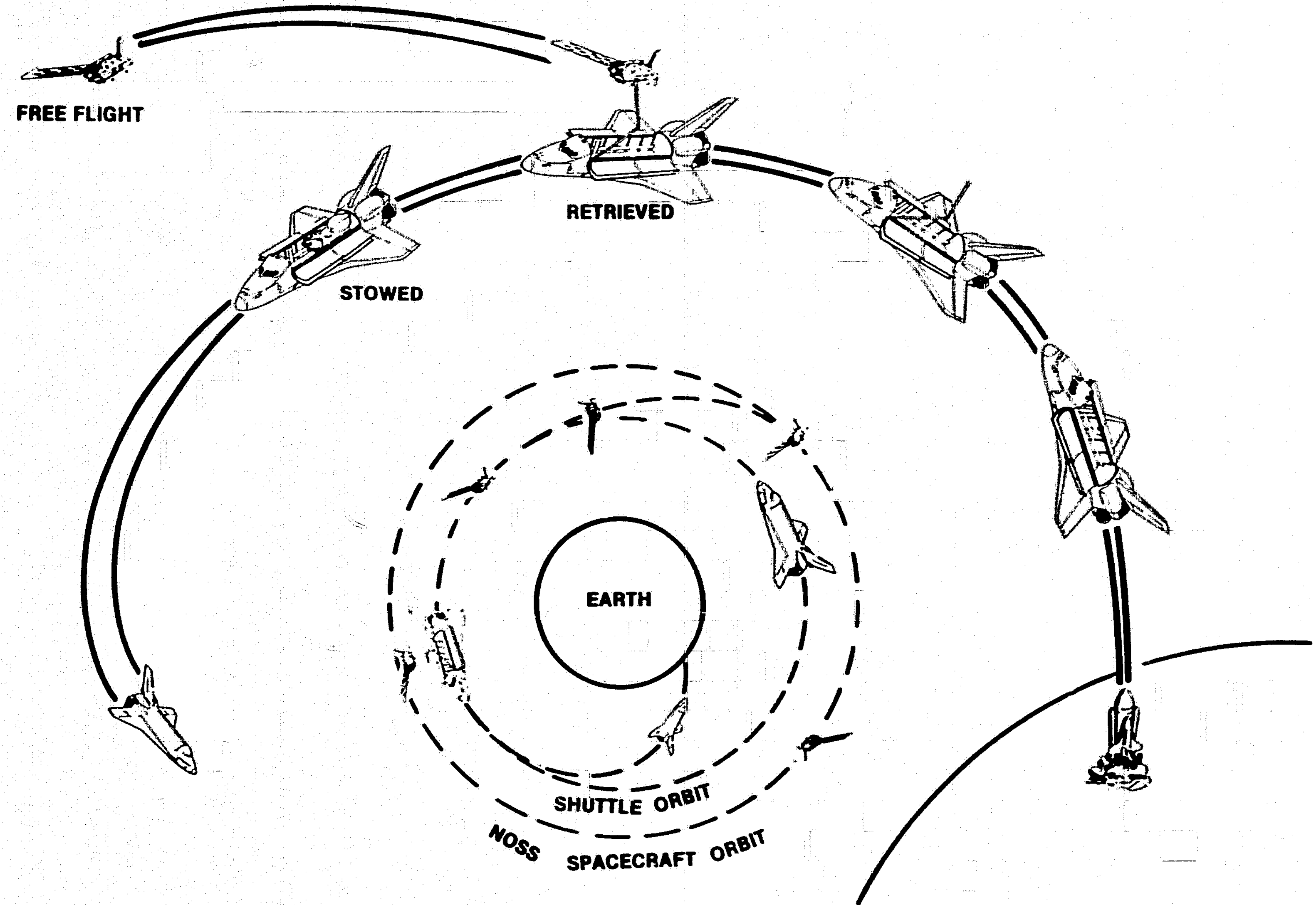


Figure 5.5-1. NOSS RETRIEVAL SEQUENCE

6.0 SUMMARY OF POTENTIAL PROBLEM AREAS, IMPORTANT SYSTEM DRIVERS AND RECOMMENDED STUDY AREAS

This section provides a summary and short explanation of items that have been uncovered during the course of study. As will be seen, they include items that are major system drivers, items that are a violation of current understandings and must be resolved, and items that require further study. They are grouped by category in the same order as the main body of this report.

6.1 STS INTERFACE

There are many areas of concern regarding the STS performance, the thermal characteristics, how clean or dirty it is, launch and landing dynamics, etc. Definition of STS payload accommodations interface is still evolving. During this study, some problems unique to the NOSS mission were identified and are addressed in this section.

6.1.1 ACS LIMITATIONS

The STS Payload Integration Plan states that a spacecraft ACS system may not be activated while the spacecraft is attached to or within 200 ft. of the shuttle. It also states that the spacecraft must induce its motion ($\Delta V \sim 1$ ft/sec) away from the STS, but that thrusters cannot be used until there is a separation of 10 nautical miles (45 minutes).

For a large spacecraft perched on the shuttle RMS, it is difficult to see how this motion can be induced without the use of small thrusters and without the aid of an active ACS. In addition, should the release occur without the ACS activated, the spacecraft could tumble into the shuttle before adequate clearance was achieved. Separation maneuvers must be resolved with the STS office.

6.1.2 INDETERMINATE MOUNTING

For a large spacecraft, the four trunnion mounting scheme described for NOSS seems to be the most logical. This, however, leads to loads on the

spacecraft induced by the flexing of the shuttle. The extent of this loading, or problems caused by it, are unknown at this time; but it is clear that a coupled loads analysis must be done very early in the spacecraft design phase.

6.1.3 WEIGHT LIMIT

The lift capability for the shuttle appears to be about 25,000 pounds for the 300 km, 98.6^{o} orbit. The current spacecraft design is about 14469 pounds and it is slightly less than 30 feet long. No attempt was made to decrease this weight but it probably could be reduced to the point where two of the spacecraft could be launched together, thus saving a second launch cost.

Additionally, STS requires special reviews of any payload within 3000 pounds of the weight limit. Hence, the STS review cycle can be minimized with resulting cost savings if the launch mass stays under this limit.

6.1.4 RETRIEVAL

Retrieval is the single most important driver in the system design, primarily in the propulsion area. More than half the size of the propulsion system is due to this constraint, and it also has significant effect on all deployment mechanisms.

6.1.5 UMBILICAL

The only ASE required for this spacecraft design is some type of plug puller, to remove the umbilical from the NOSS prior to its removal from the cargo bay. This will have to be mounted on the shuttle sill in some fashion and was not addressed in this study.

6.2 TDRSS INTERFACE

6.2.1 S-BAND SINGLE ACCESS (SSA) DATA RATE

The SSA data rate is currently limited to 3 MBPS, whereas it previously was 6 MBPS. The limitation, however, is not in the TDRSS spacecraft but in the implementation of the demodulators at the White Sands facility. To dump two tape recorders simultaneously from NOSS, will require 2.66 MBPS on both the I and Q channels. Either the additional demodulator must be installed, NOSS must install its own demodulators and interface at IF with TDRSS, NOSS most dump the tape recorders sequentially, or NOSS must switch from S-Band to K_u-Band. None of these presents a fundamental problem, but a trade-off study must be done to choose a solution.

6.3 LAMMR

This instrument presents a real design problem to all areas of the spacecraft. Other items relating to this instrument appear later in this section under the ACS and the mechanical areas.

6.3.1 LAMMR ANTENNA STOWAGE

From trade-offs done during this study in the layout of the spacecraft, a major effect emerged. If the antenna is stowed "horizontally" in the shuttle (chosen for this design) then the resultant spacecraft structure and deployments are considerably simplified. If the antenna is stowed "vertically" in the shuttle, the spacecraft design becomes considerably more complicated.

6.3.2 FLEXIBLE BODY ANALYSIS WITH PROPELLANT SLOSH

The residual unbalance of the LAMMR will impart a cyclical torque to the spacecraft probably about all three axes. This leads to the necessity for a study of the spacecraft flexure under the excitation imposed by the LAMMR. For this design, it is likely that the spacecraft body can be considered rigid with the propellant slosh centrally located and the only flexible parts being the solar array, the high gain antenna, and the LAMMR itself.

6.3.3 ACTIVE TORQUE COMPENSATION

The concern over the torques imposed on the spacecraft by the LAMMR also indicate that possible "active" means be studied to cancel or eliminate this residual torque.

6.4 SCAT

6.4.1 ANTENNA GROUND PLANE

The SCAT antennas were originally designed to be deployed. In the attempt to minimize deployments, the antennas were regidly mounted to the Earth viewing side of the S/C. Since they are fairly high gain antennas, this should not present a problem but it must be tested and evaluated if this type of mounting is to be acceptable.

6.4.2 ANTENNA PATTERN

The two antennas parallel to the velocity vector of the spacecraft produce the two beams perpendicular to the sub-satellite ground track. These beams cannot be at 90° to the track, since doppler information is used to determine the location of the cells. Hence, these must be at some other angle, the most likely candidate being 75°. This information was obtained too late in the study for the changes to be incorporated, but there is room to split the two antennas into a "vee" shape with minimum effect on the spacecraft design

6.5 CZCS

6.5.1 CONTAMINATION

This is the only instrument sensitive to contamination and may need a cover. Due to the large amount of room around the instrument, a dome-like cover could easily be installed that could be jettisoned in the shuttle orbit. Again, this is not included in the current design but is a simple addition. Contamination on the return flight should not be a problem because the instrument can be cleaned.

6.5.2 COOLER

The cooler for the CZCS requires a 101° circular field-of-view. In its present location, part of one spacecraft trunnion impinges on this FOV. If "scuff plates" are required by the STS for retrieval, the blockage will be increased. In either case the CZCS will have to be moved. Again, there is ample

room to accomplish this movement so the spacecraft design remains the same.

6.6 GPS

6.6.1 ACCURACY

The GPS system developed for LANDSAT-D has been assumed for NOSS. It uses two frequencies, however, to compensate for ionospheric effects and the high resolution (classified) code for precise position determinations ($\approx$ 10M). NOSS only requires $\approx$ 400M accuracy and the possibility of a much simpler system perhaps using only one frequency and the low resolution (unclassified) code should be investigated.

6.7 POWER

6.7.1 SOLAR ARRAY

In the NOSS time frame, it appears that the array chosen for this spacecraft, $\approx \frac{1}{2}$ of the PEP array, offers the best available performance features with minimal technical risk. This design will be flight tested in FY83. All others, including the FRUSA designs, are presently being developed in versions too small for this application.

6.8 PROPULSION

6.8.1 LIQUID VS. SOLID

The retrieval sequence has the NOSS moving down to the STS orbit prior to the STS arrival. Depending on the amount of time required or on potential STS launch problems, the spacecraft must be capable of maintaining itself in low orbit for up to two weeks. This requires a solar array. If the NOSS appendages must be retracted or jettisoned prior to return from the mission orbit ($\approx$800 km), a separate array providing keep-alive power must be available. As this is several hundred watts of power, this is an expensive item. To eliminate this, the return orbit transfer must be accomplished with the solar array deployed, hence, low thrust levels, hence a liquid system.

This then enables the initial deployment to take place in the low STS orbit if the same system is to be used for the ascent.

Should the retrieval requirement go away, other factors will have to be evaluated to decide on either a dual solid or a liquid system for ascent, but with this retrieval requirement, the liquid systems seems best.

6.8.2 TRANSFER MODE

Several orientation (for transfer from the STS orbit to the mission orbit and back again) were examined along with the options of multiple 45 pound thrusters versus a single 240 pound thruster.

The idea of the 90° rotation wherein the Earth viewing face is moved perpendicular to the velocity vector and using 4 (45 lb) thrusters seems best in that it uses only 4 and provides redundancy. Any other scheme either requires multiple large thrusters (2 to 4) for redundancy or even a larger number of the smaller thrusters (8). Potential failure modes may require that additional thrusters be added, however, this design proposes the minimum number of thrusters. For the backup mode of transfer with only two (45 lb) thrusters, additional propellant through-put testing will have to be done.

Additionally, the four smaller thruster configuration is more tolerant of a displaced CG and, hence, considerably easier to control during the transfer.

6.9 ACS

6.9.1 DISTURBANCES

Several items on this spacecraft will cause significant disturbances to the ACS. Several have already been discussed under the LAMMR section (the flexible appendages and fuel slosh). Three additional ones should be **studied** for both the mission orbit and the STS orbit. These are the gravity gradient forces, the drag of this large structure, and solar pressure. A first look has been taken at these, but it should be done in greater detail.

6.9.2 EARTH SENSOR

This item is not strictly required to accomplish the NOSS mission, but offers several advantages as well as functional backup and was included for that reason.

6.9.3 ANALOG BACKUP CONTROLLER

This item is another result of retrievability. For ascent, the mission would be aborted if, in the low STS orbit, the computer failed. Since this is a candidate for on-orbit failure and due to the loss of precision the spacecraft would be useless, some means should be available to control the spacecraft and bring it back to the STS orbit. A simple analog controller seems best suited for this task.

6.9.4 STAR TRACKER

During the layout phase of this design, the star tracker was removed from the ACS module to provide clear fields-of-view. Subsequent to this, other components were moved and it now appears feasible to have the star tracker in the ACS module. It has a better FOV in its current location but it would still work in the ACS module. This was realized too late in the study for this change to be made.

6.9.5 SENSOR POINTING ATTITUDE DETERMINATION

The critical attitude angle (0.03°, 3σ) is that for the LAMMR pointing angles. Spacecraft flextures may cause deviation between the star tracker and LAMMR antenna axes alignments which must be determined. The study has not addressed this issue in detail.

6.10 C&DH

6.10.1 TAPE RECORDERS

There does not appear to be a tape recorder in the 10^9 bit range

currently available. There was to be a NASA standard 10^9 bit recorder developed by Odetics, but work was halted in the engineering phase and is probably several years and many dollars away from being developed. No development is now going on. Beyond that, are only options for modifying other existing tape recorders. This area will be a problem and requires additional study.

6.10.2 INSTRUMENT DATA AUTONOMY

The design of the NOSS instrument data telemetry streams will provide information necessary to perform primary data processing directly upon receipt of the instrument data. This implies that sufficient time, spacecraft attitude and orbit data are contained within the flight data telemetry. It appears that this information must be added to the basic instrument data following receipt of the data by the HDR equipment. A detailed effort will be required to determine the precise manner and the interfaces required to satisfy the data processing accuracy. Each instrument presently outputs data in a serial mode after some processing delay; therefore, the ability to externally merge the necessary header data and the resolution possible for each instrument need additional study.

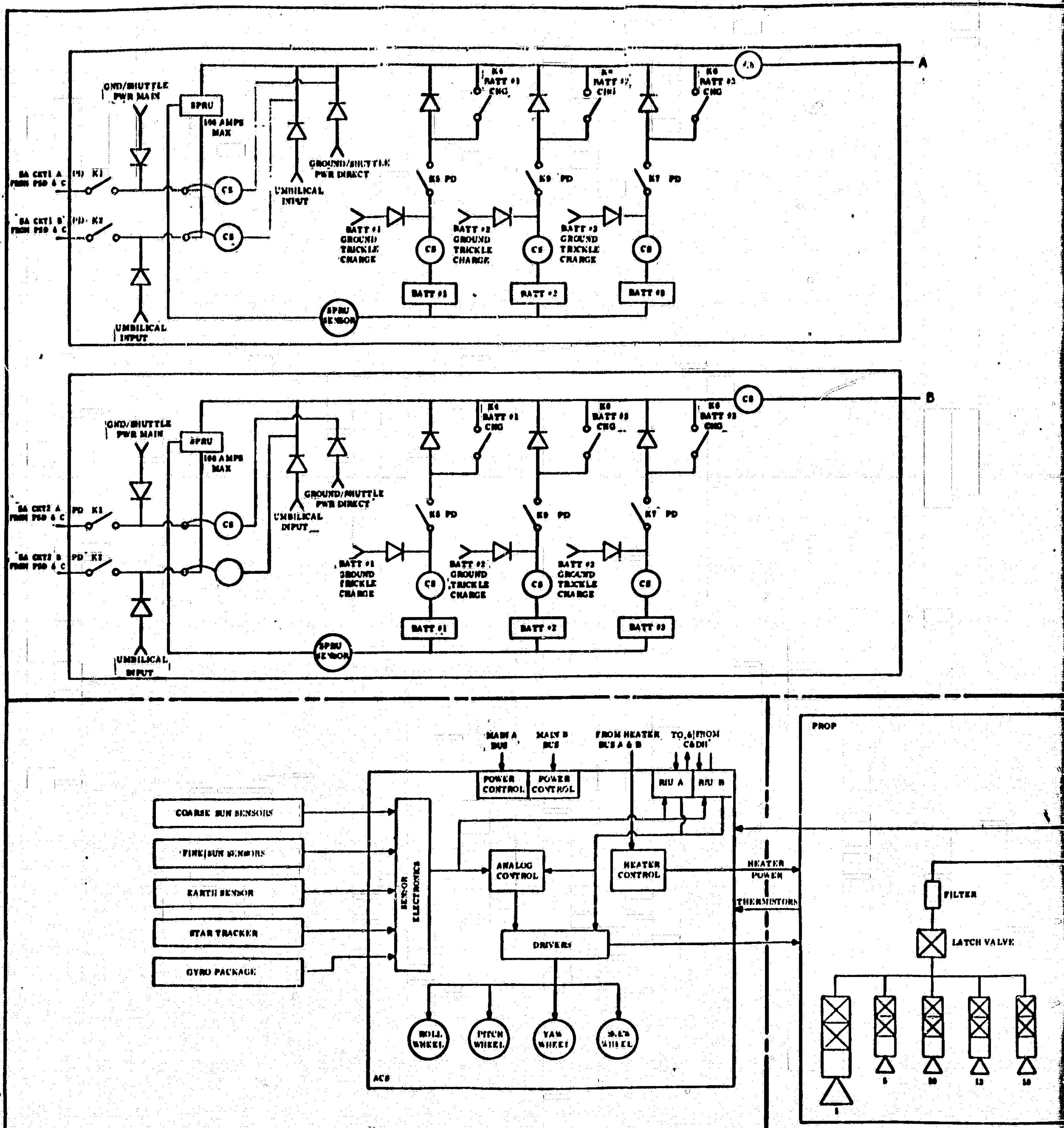
7.0 DRAWINGS AND SYSTEM BLOCK DIAGRAM

The engineering drawings and the system block diagrams generated for this study are contained in this section. The drawings are reduced from the 1/8 scale used to generate the drawing and the system block diagram (Figure 7-1) is included as a fold-out to enhance visualization of the inter-relationship between the subsystems and instruments.

The drawings which are contained in this section are listed below:

Figure 7-10 NOSS Spacecraft - Deployed Configuration - Side View - Shady Side

Figure 7-11 NOSS Spacecraft - Deployed Configuration - End View



FOLDOUT FRAME

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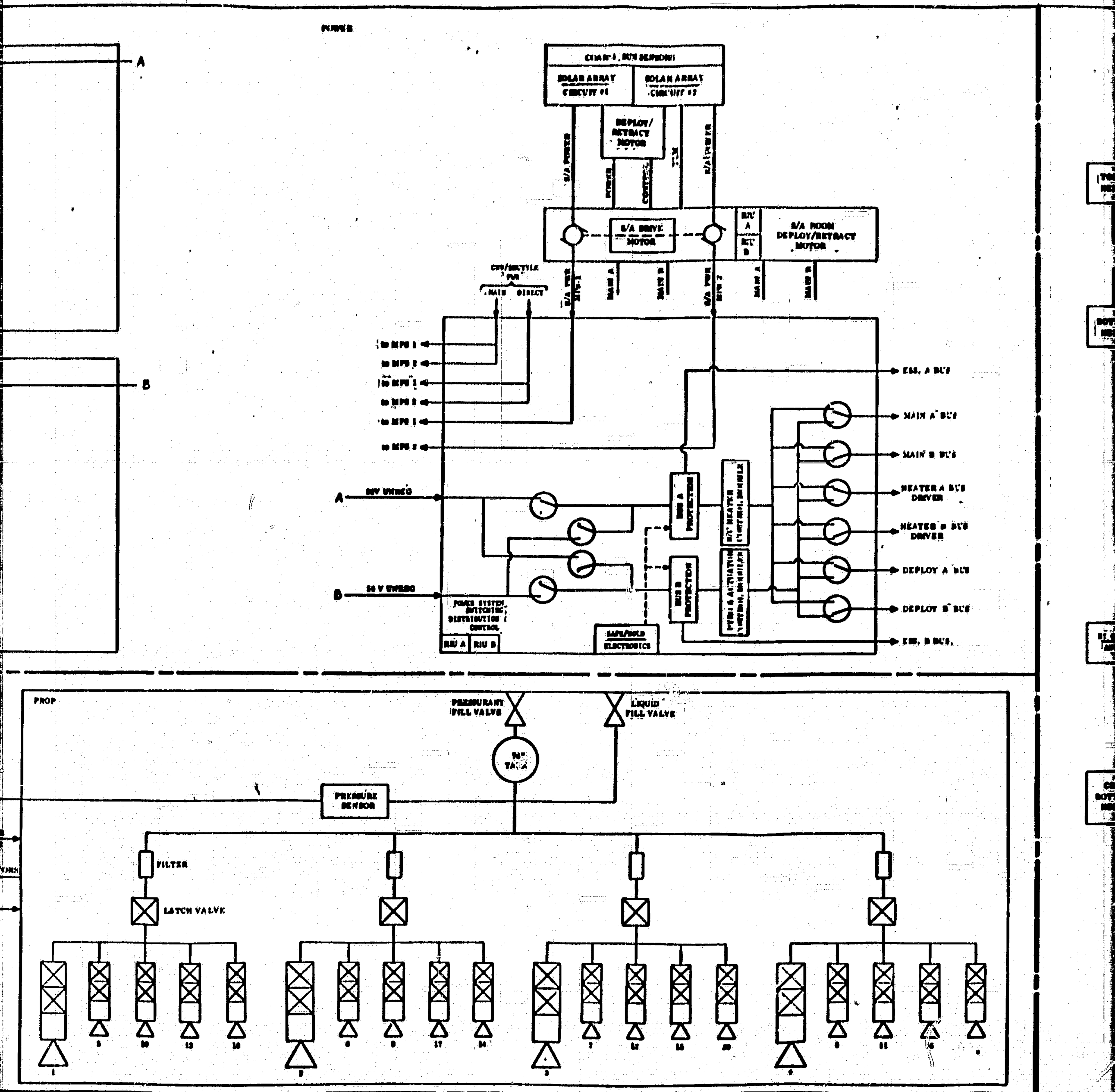

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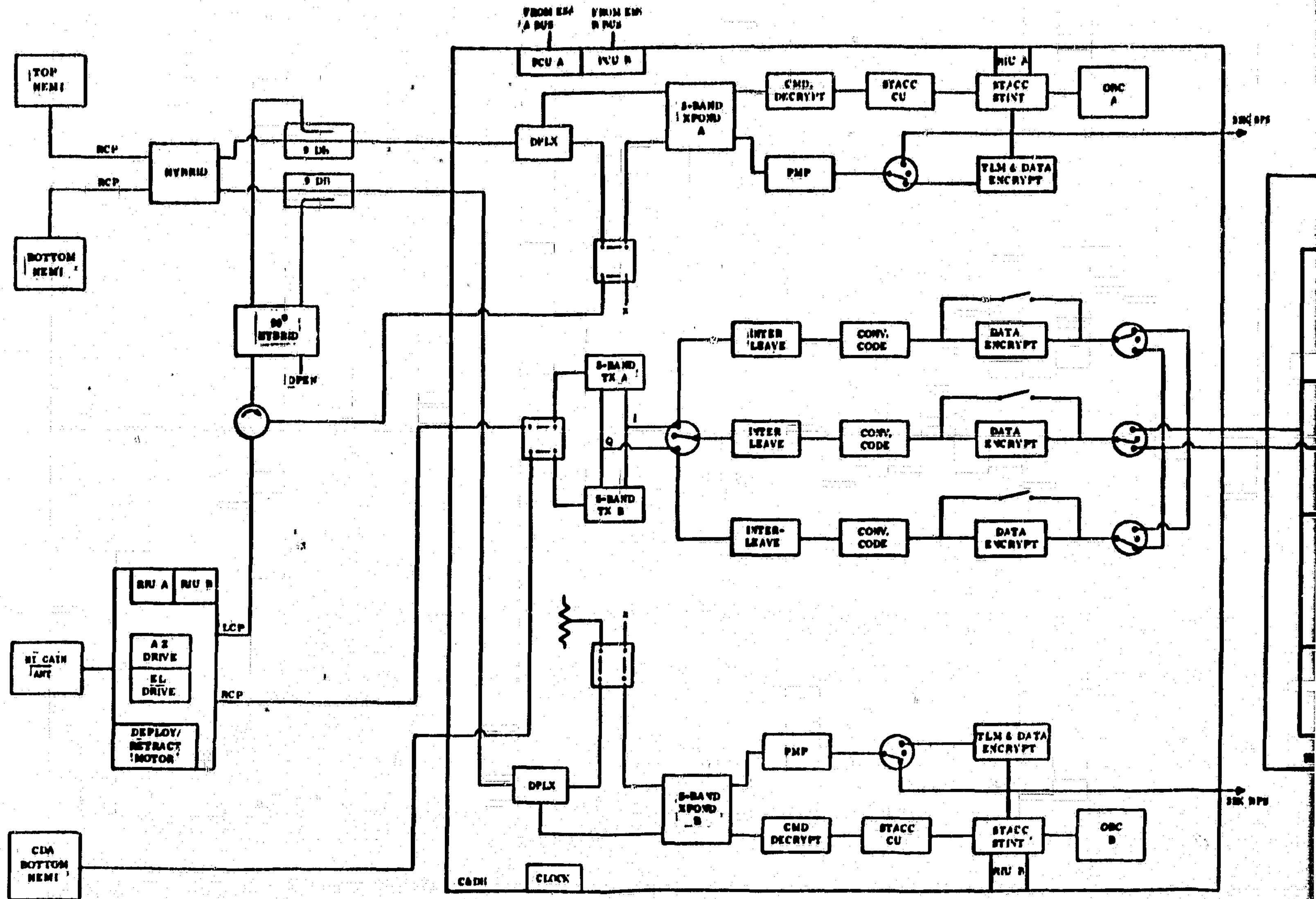

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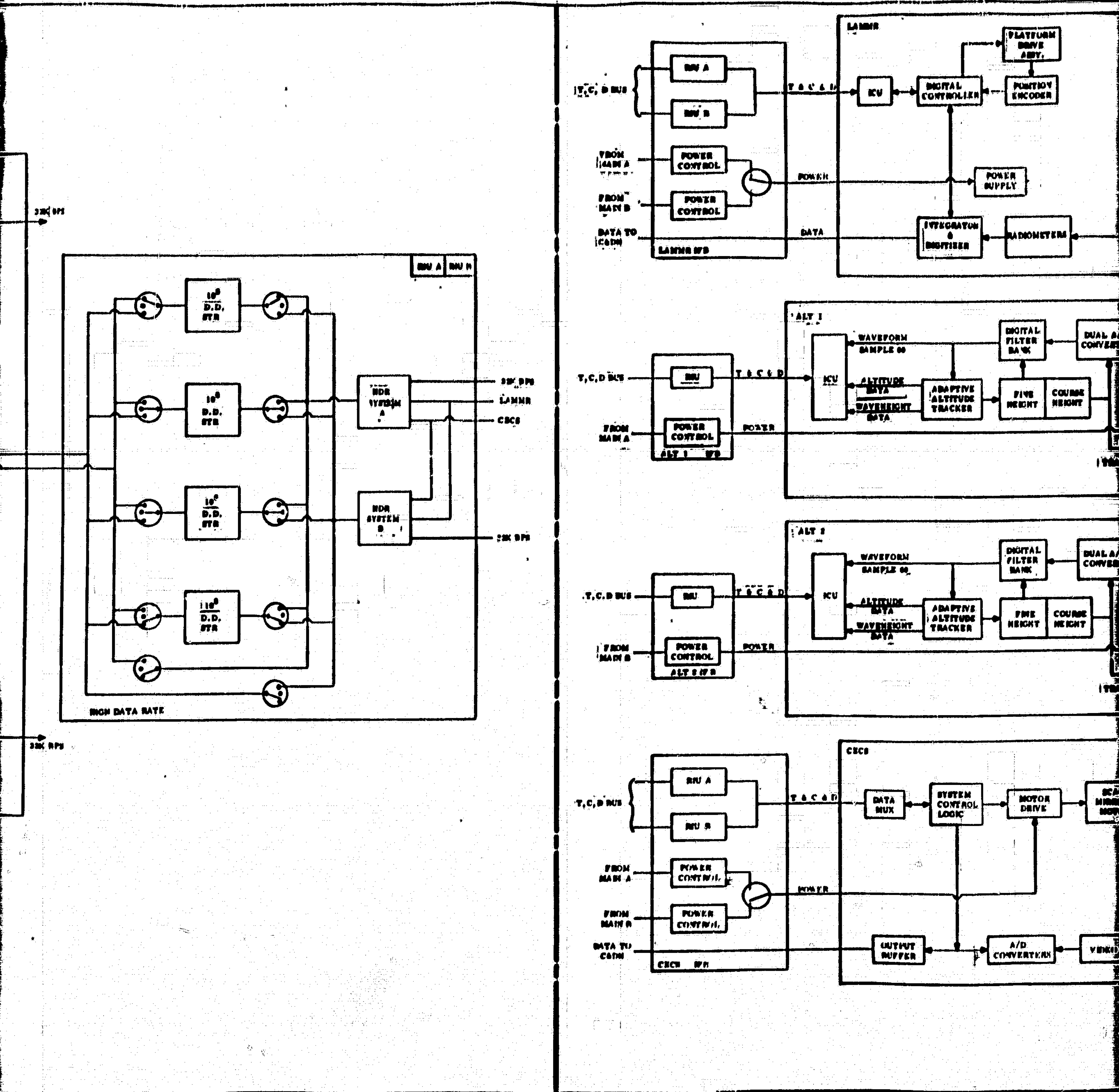

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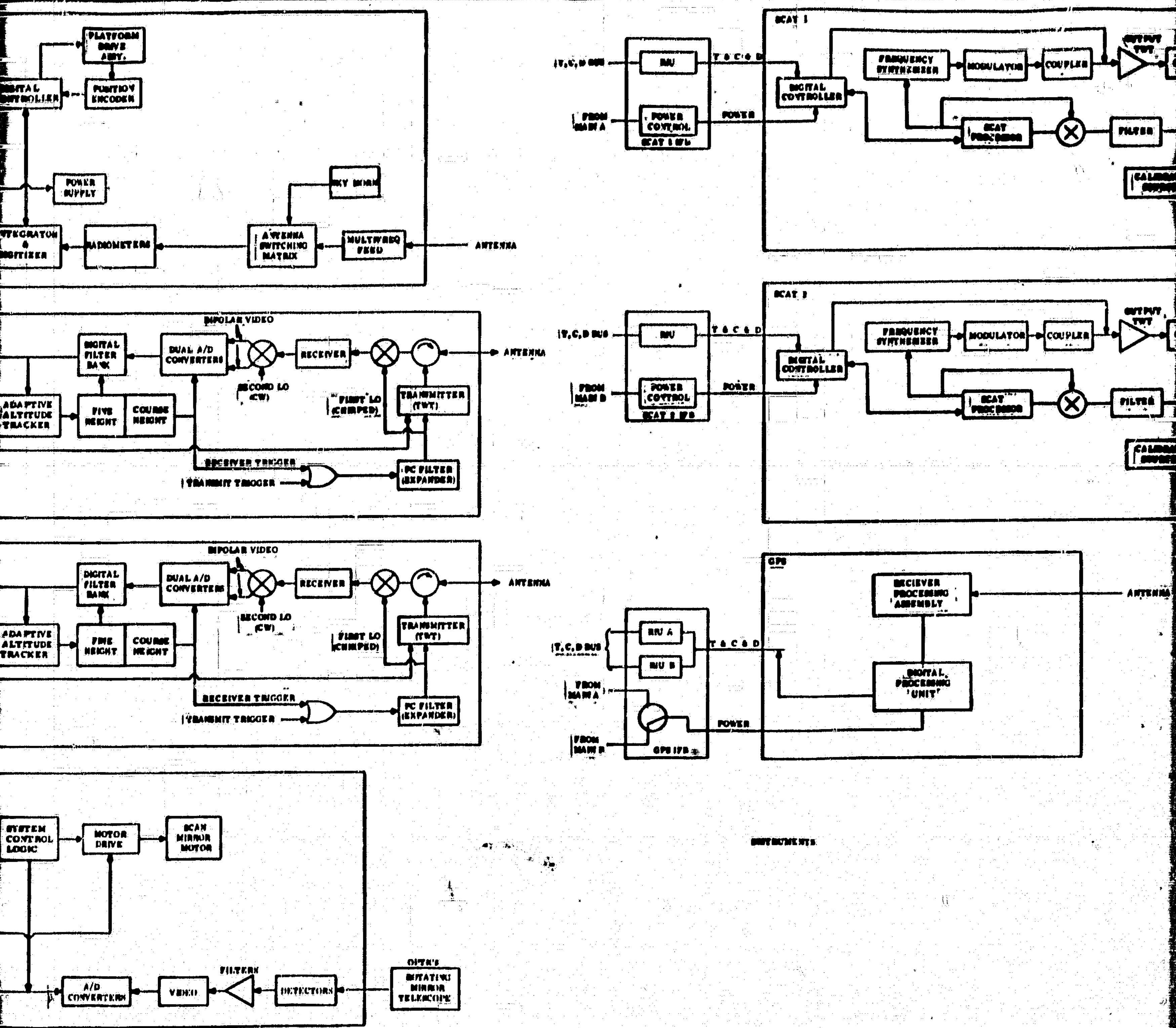

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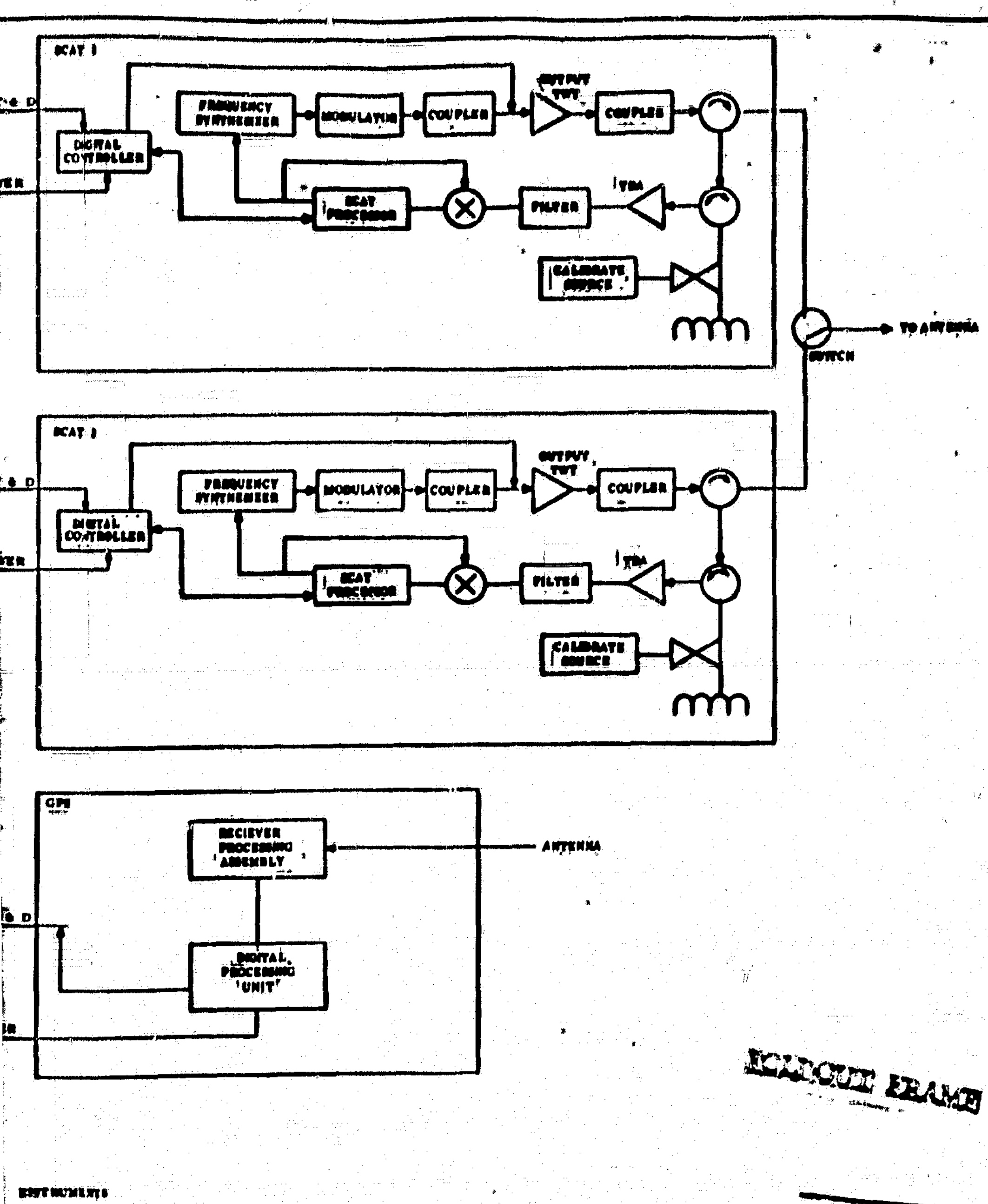
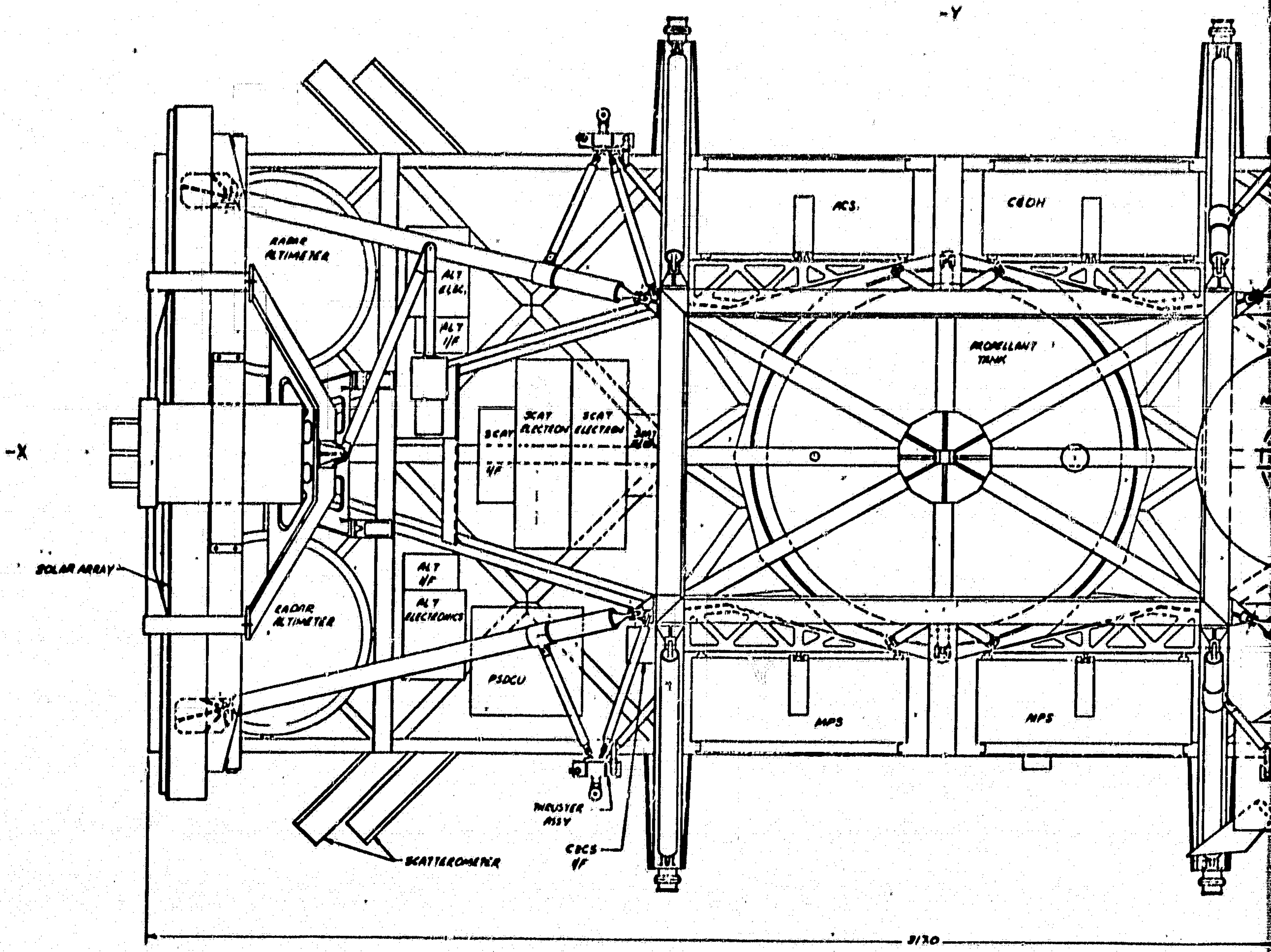


FOLDOUT FRAME 6

FIGURE 7-1
NOSS SYSTEM BLOCK DIAGRAM


-Y
-X
+Y
SOLAR ARRAY
RADAR ALTIMETER
ALT ELEC.
ALT I/F
SCAT I/F
SCAT ELECTRON
PSDCU
ALT ELECTRONICS
ACS.
CEDH
PROPELLANT TANK
MPS
THRUSTER ASSY
CDCS I/F
SCATTEROMETER
3130

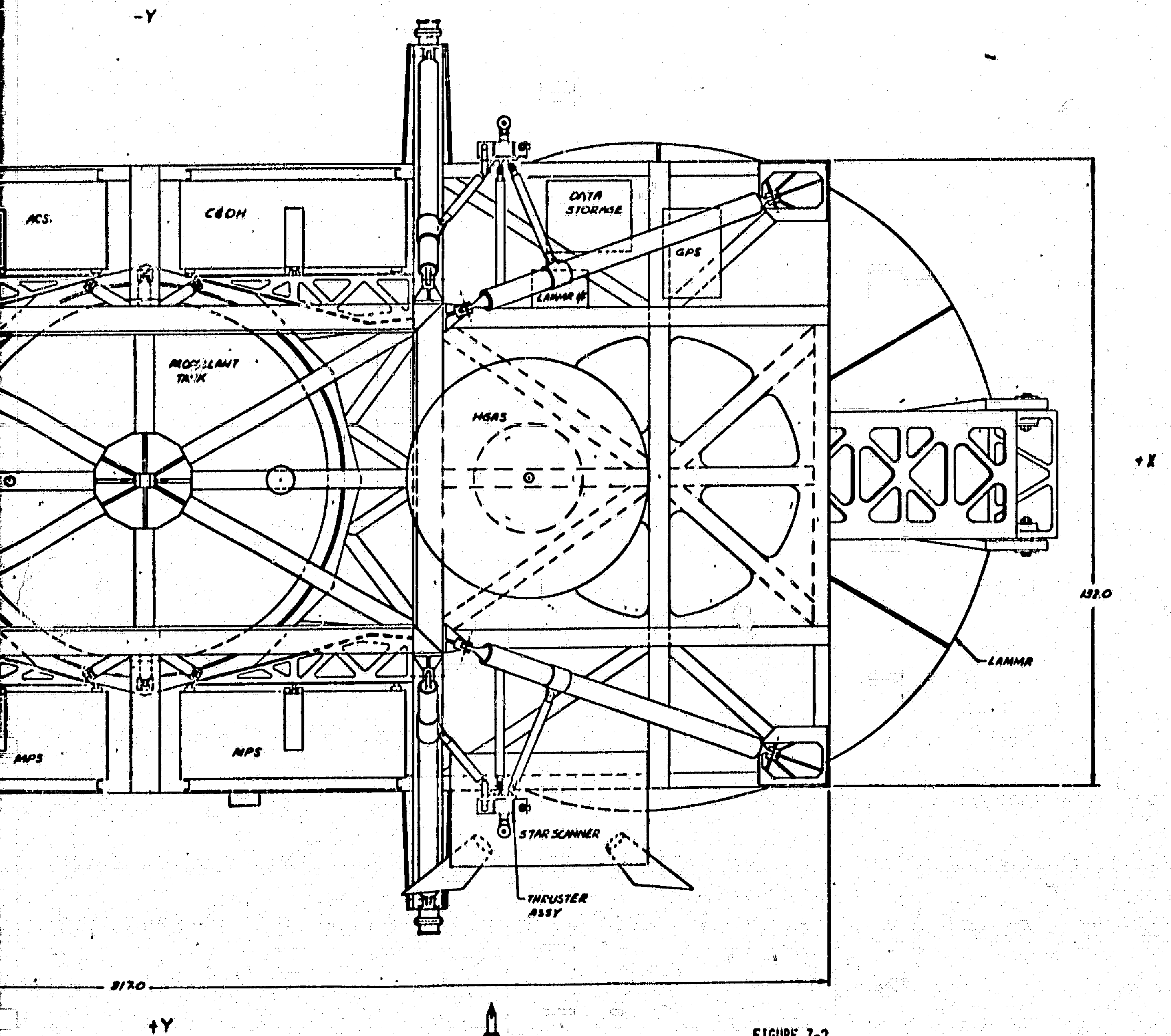
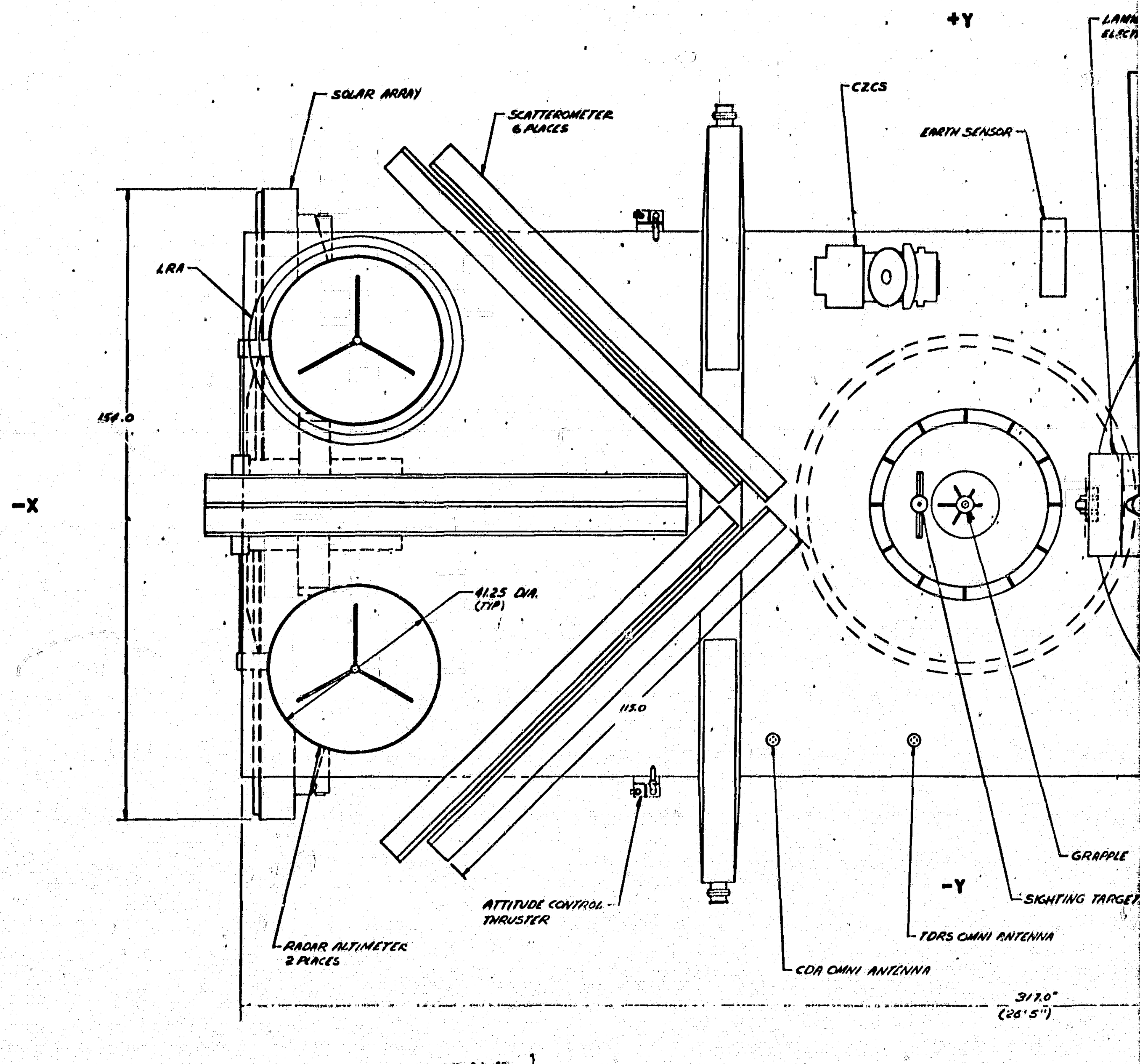


FIGURE 7-2
NOSS SPACECRAFT - STOWED CONFIGURATION - TOP VIEW - SPACE VIEWING SIDE

FOLDOUT FRAME 2



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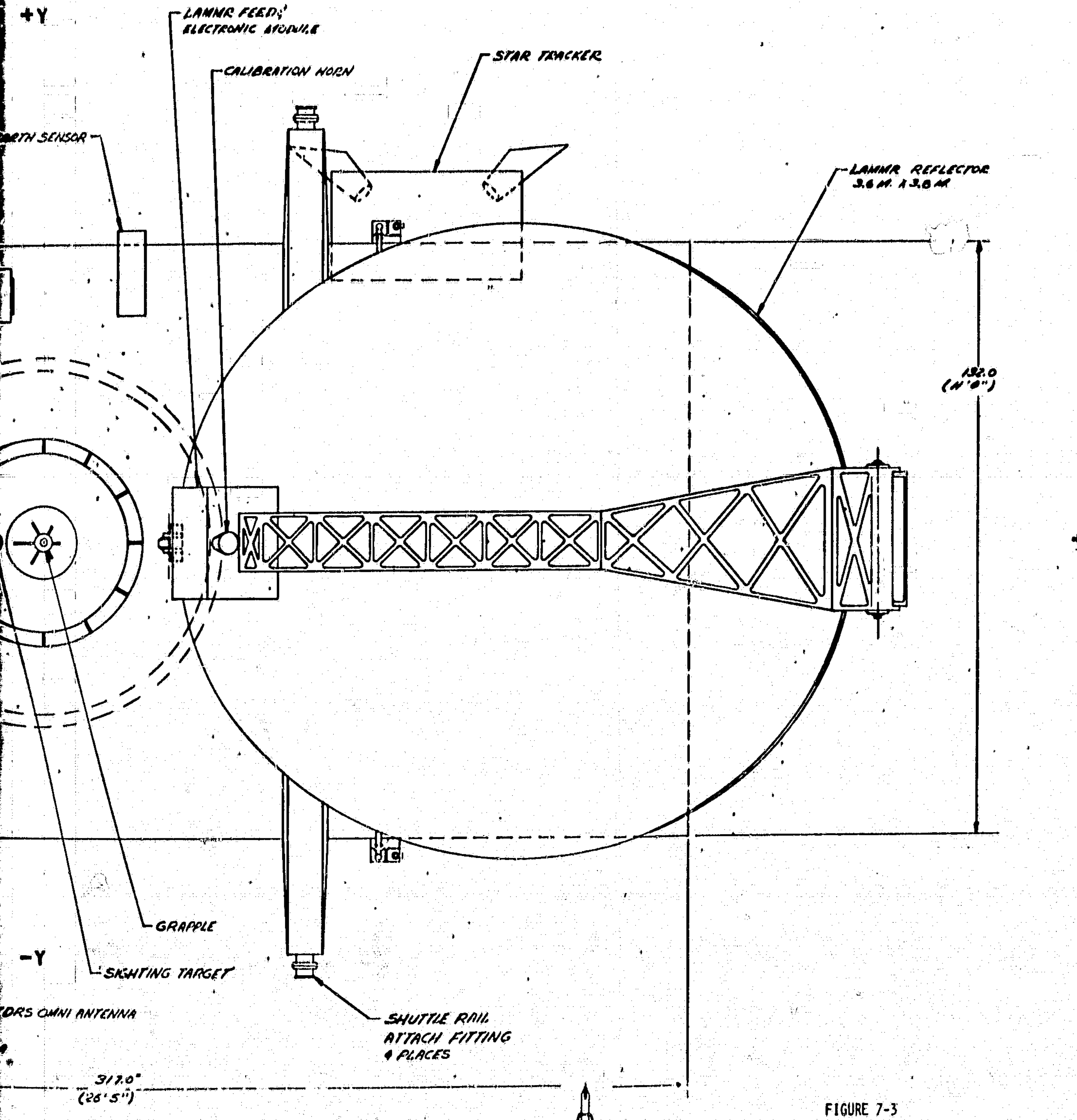


FIGURE 7-3

NOSS SPACECRAFT - STOWED CONFIGURATION - BOTTOM VIEW - EARTH VIEWING SIDE

FOLDOUT FRAME 2

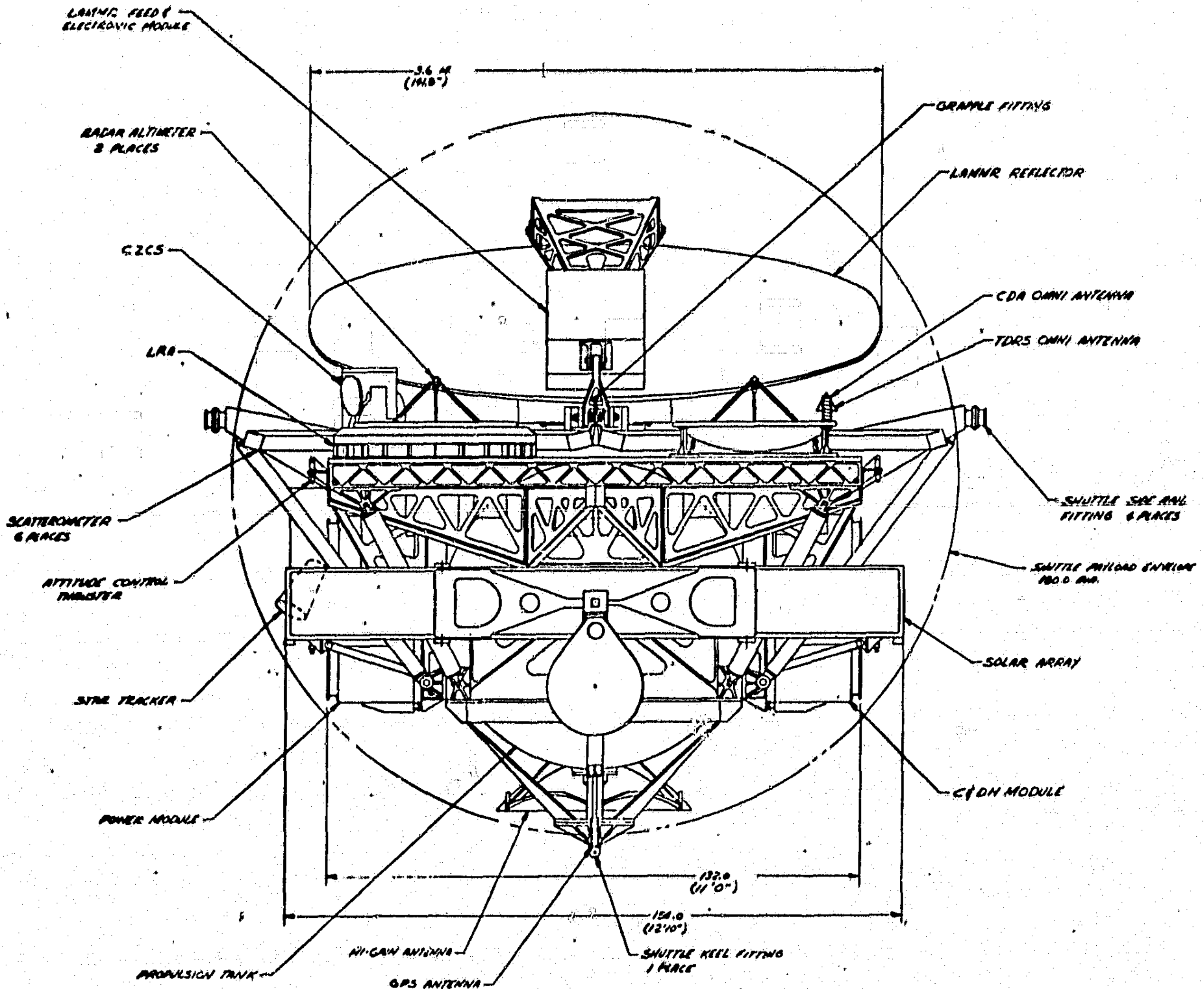
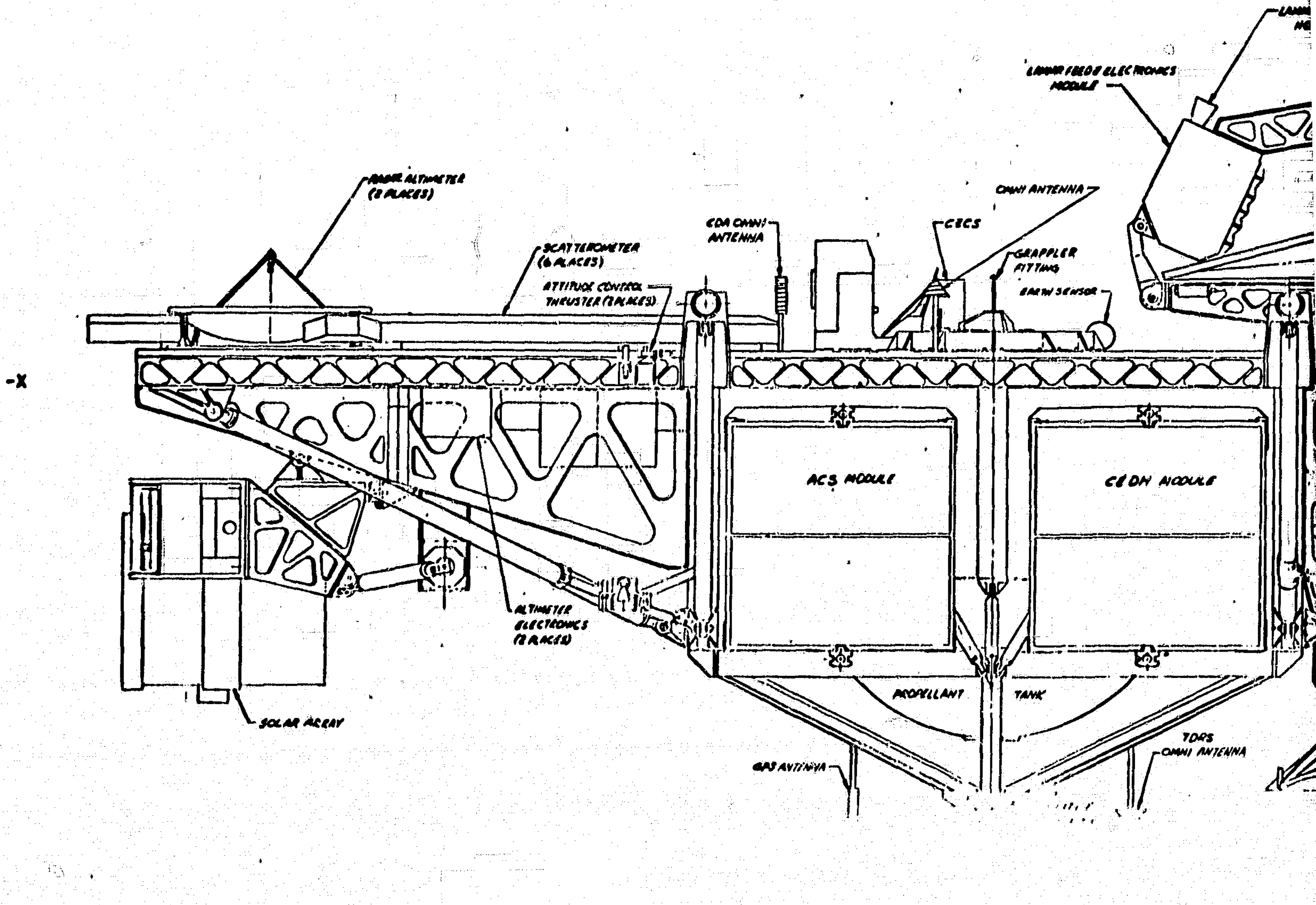


FIGURE 7-4
NOSS SPACECRAFT - STOWED CONFIGURATION - END VIEW - SHUTTLE ENVELOPE



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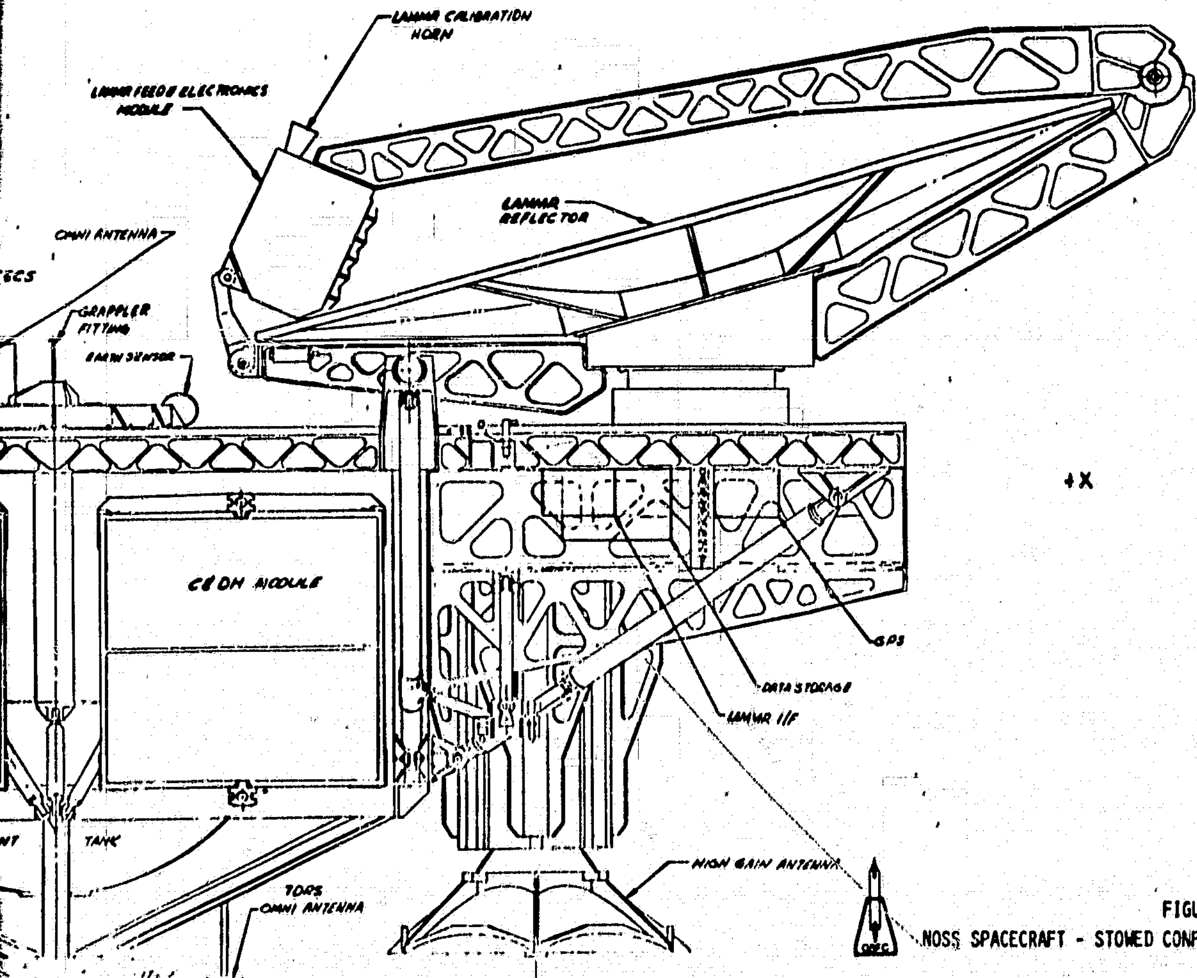
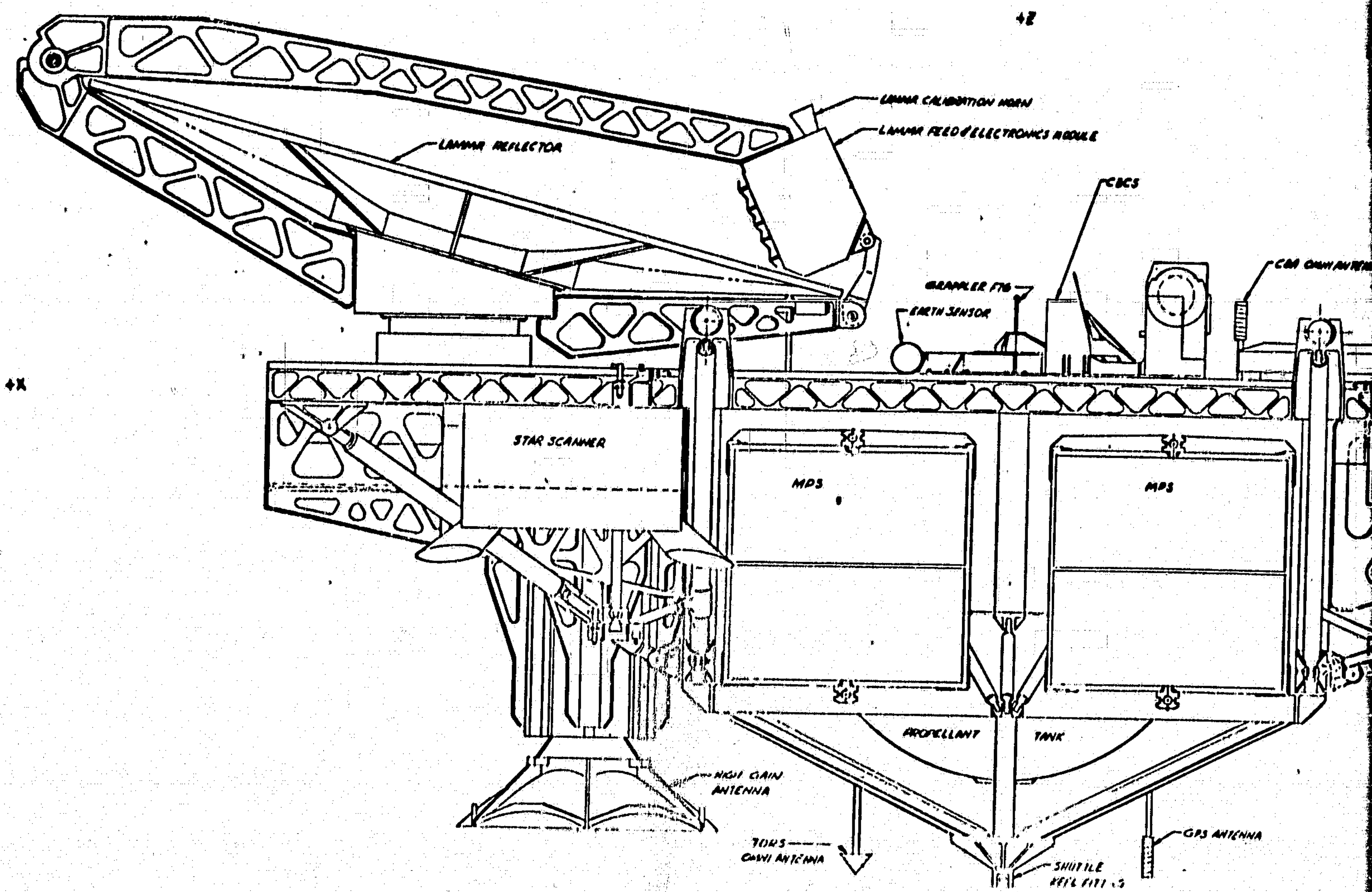


FIGURE 7-5
NOSS SPACECRAFT - STOWED CONFIGURATION - SIDE VIEW - SUN SIDE

FOLDOUT FRAME 2



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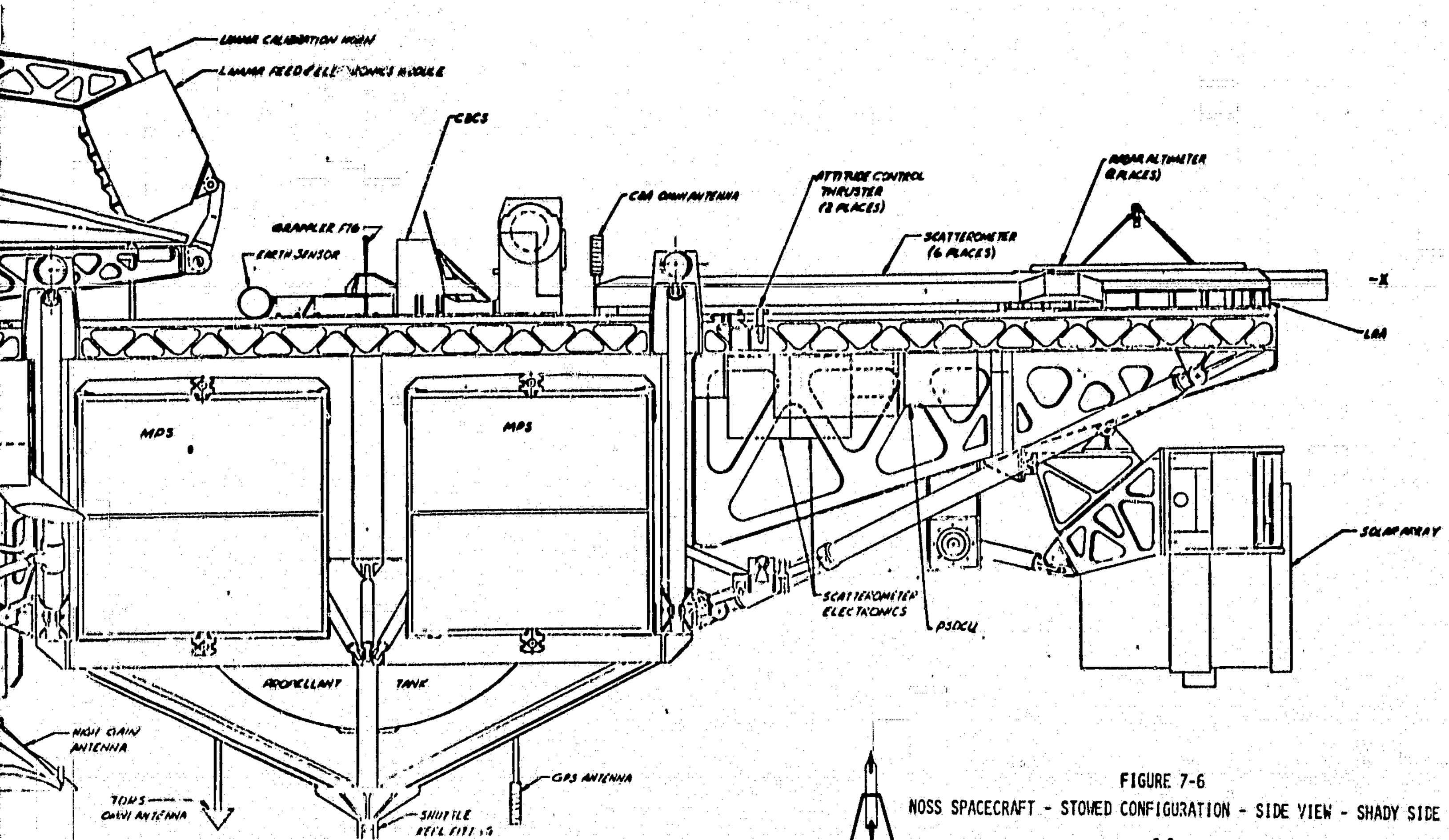

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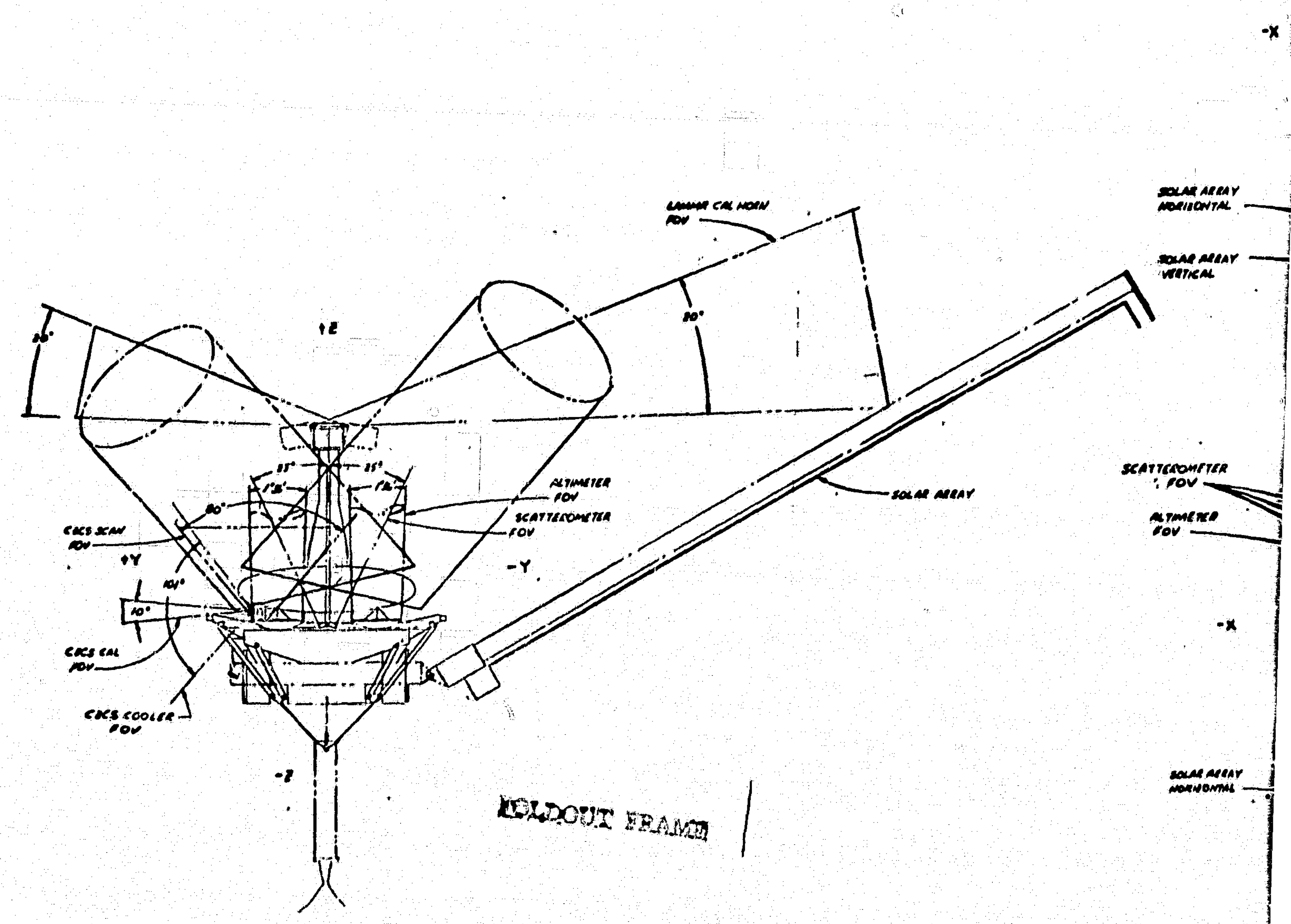
FIGURE 7-6
NOSS SPACECRAFT - STOWED CONFIGURATION - SIDE VIEW - SHADY SIDE

FOLDOUT FRAME 2



FOLDOUT FRAME

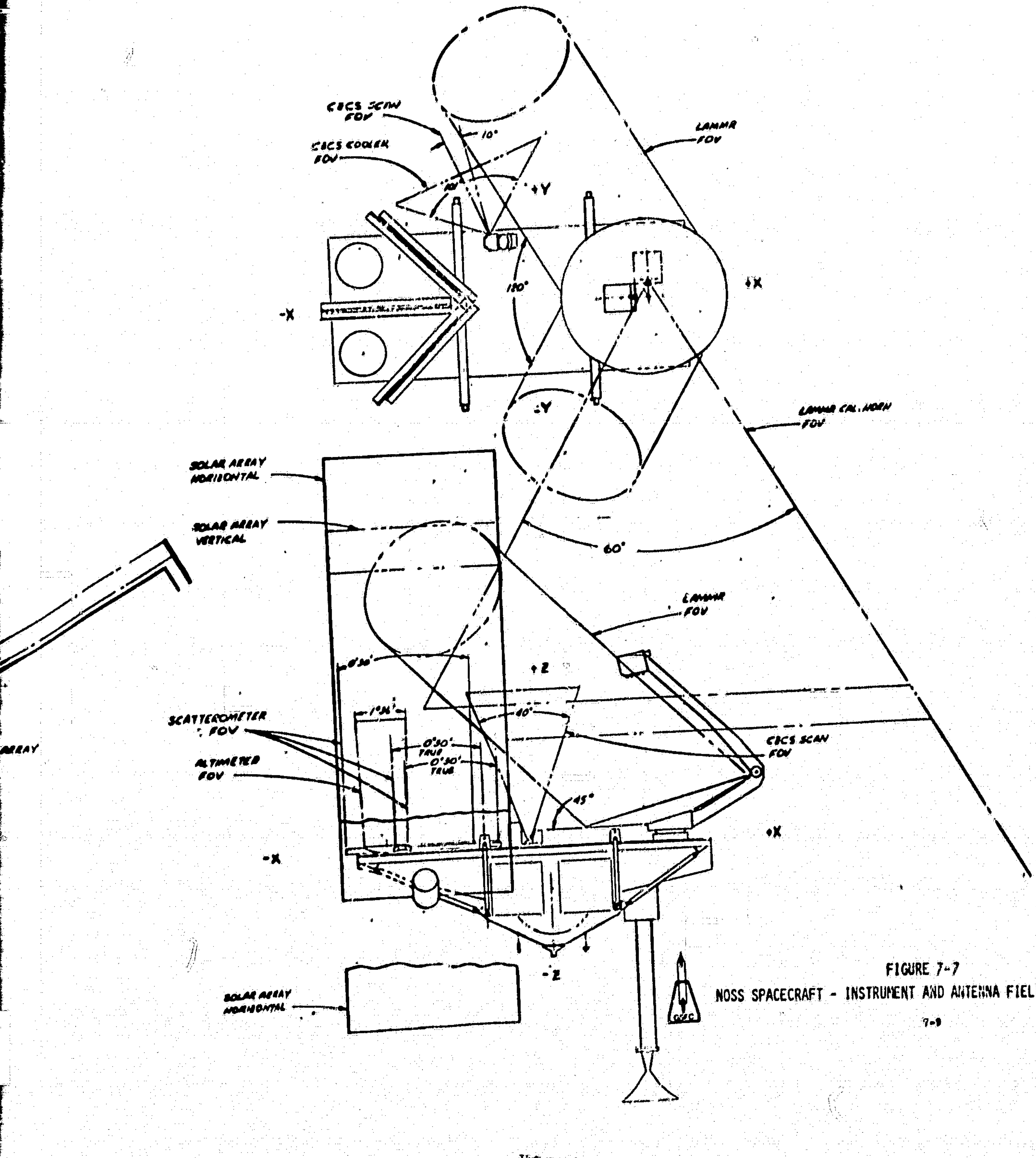


FIGURE 7-7

NOSS SPACECRAFT - INSTRUMENT AND ANTENNA FIELDS-OF-VIEW

FOLDOUT FRAME 2

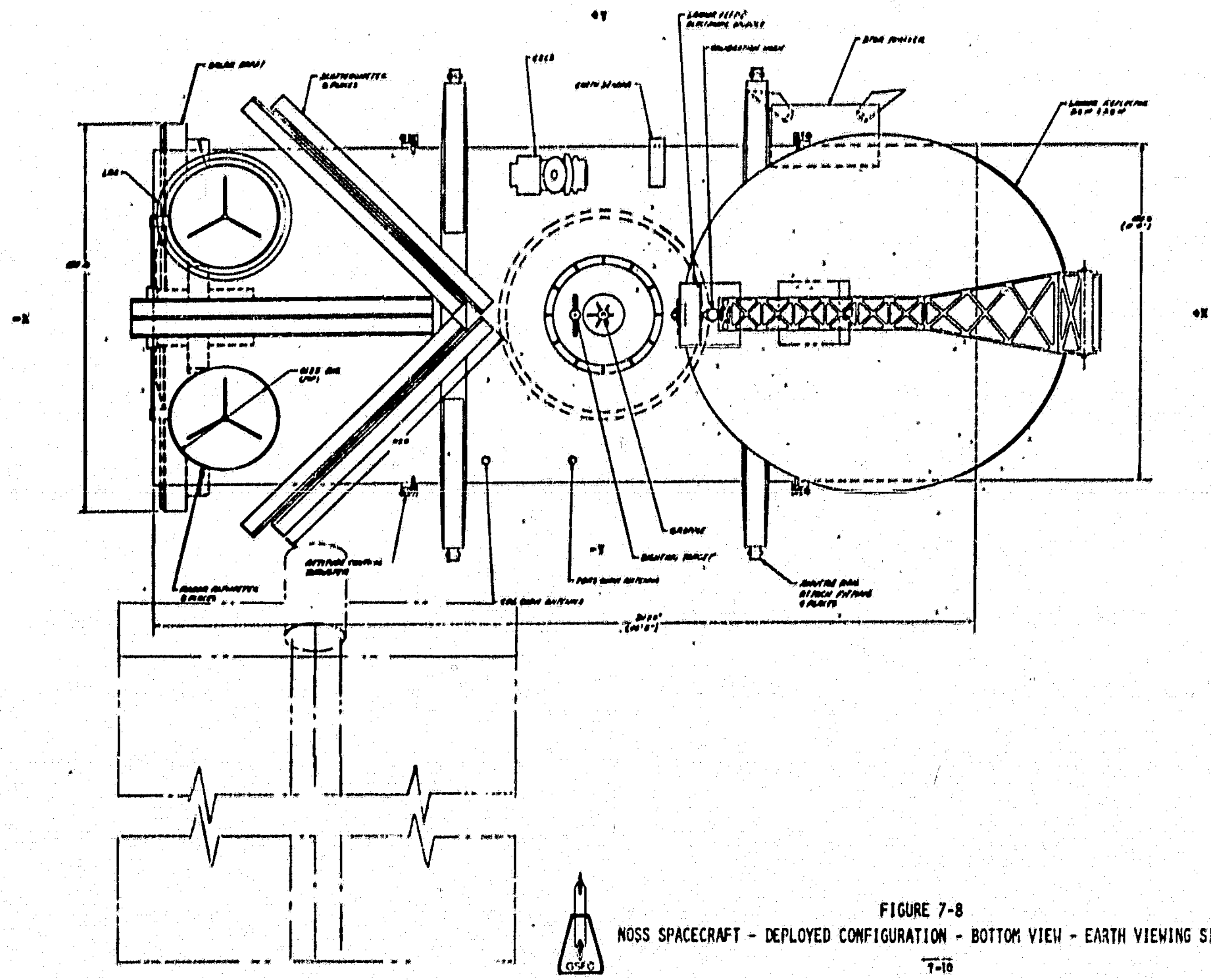


FIGURE 7-8
NOSS SPACECRAFT - DEPLOYED CONFIGURATION - BOTTOM VIEW - EARTH VIEWING SIDE

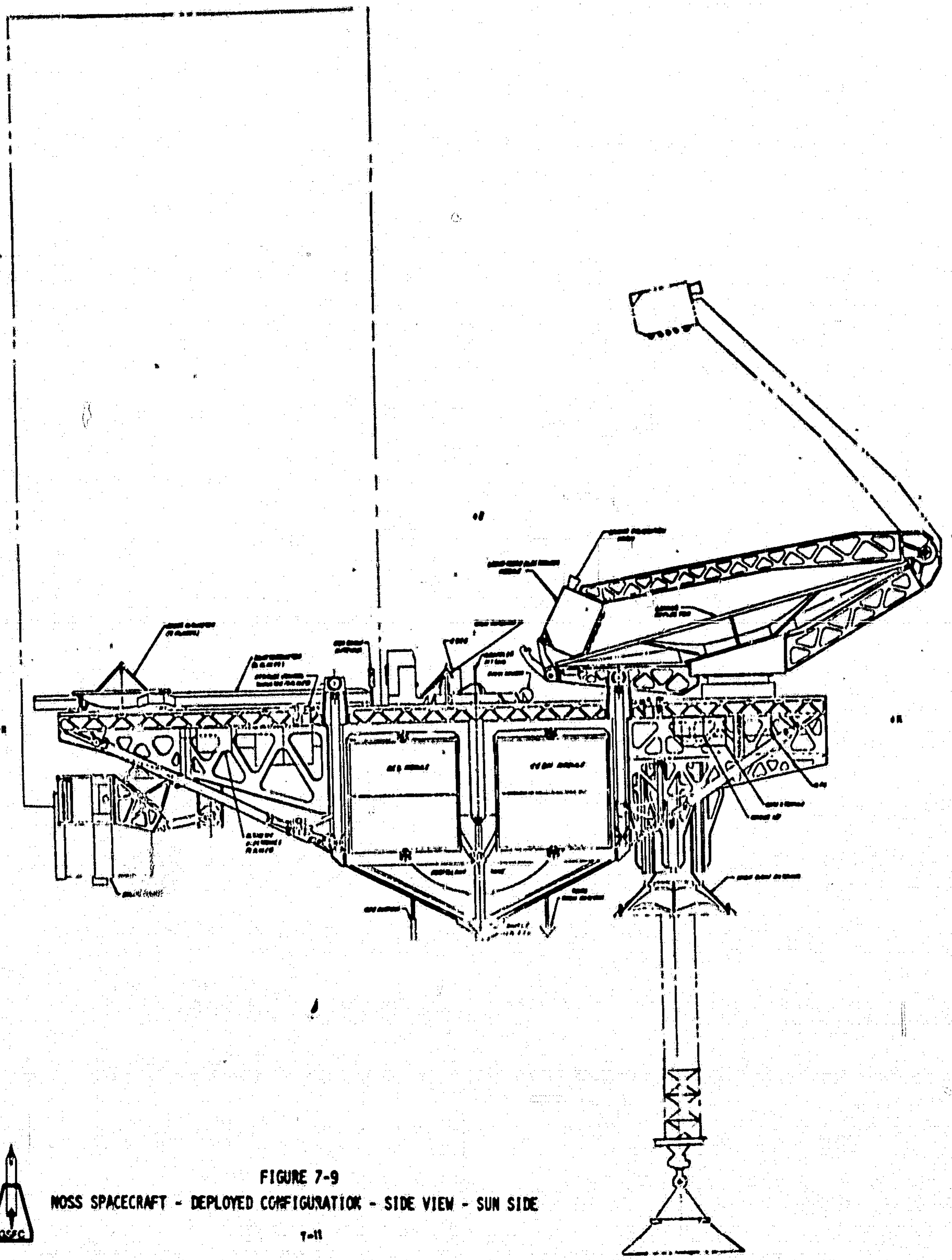


FIGURE 7-9
NOSS SPACECRAFT - DEPLOYED CONFIGURATION - SIDE VIEW - SUN SIDE

ORIGINAL PAGE IS
OF POOR QUALITY

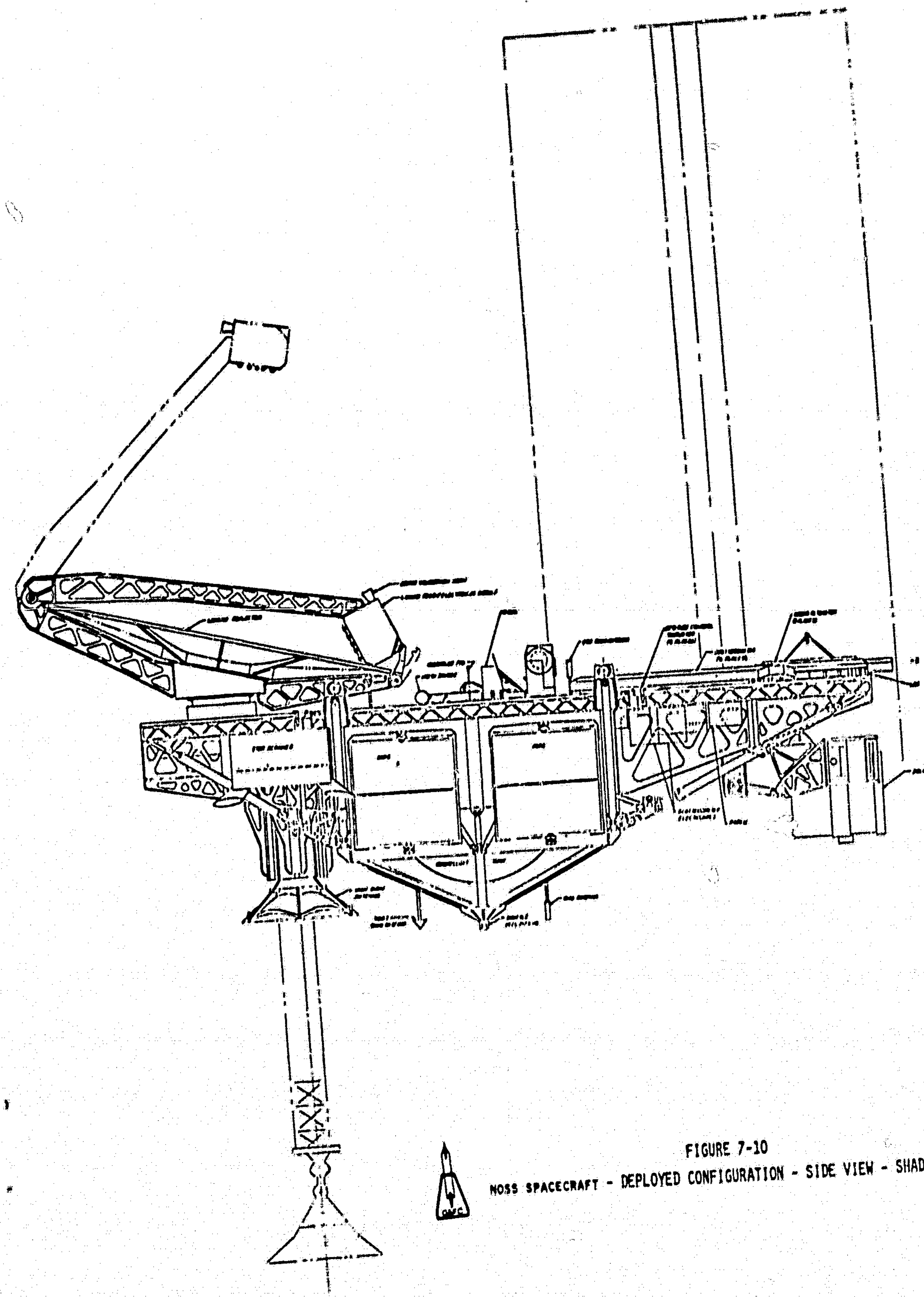
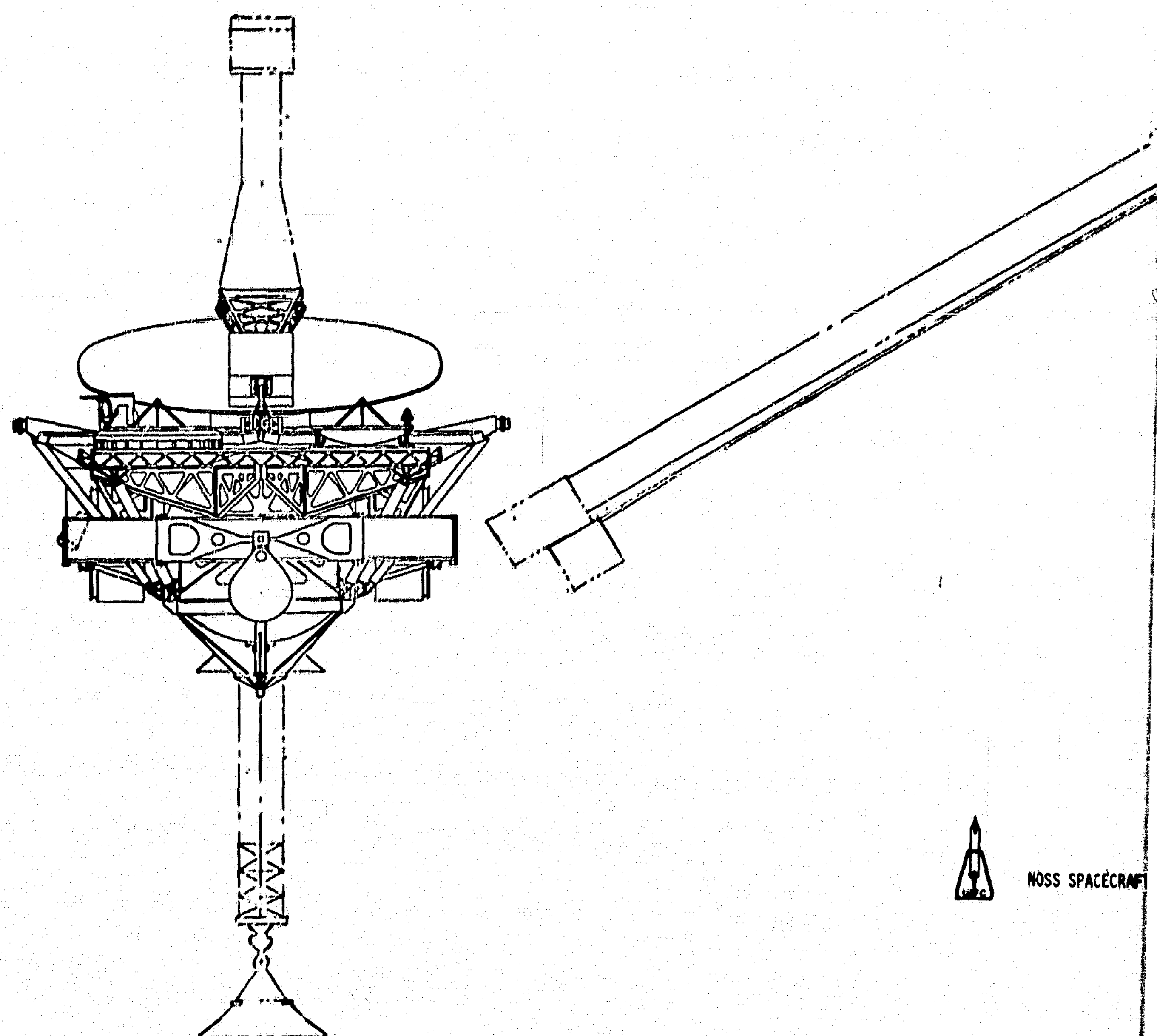


FIGURE 7-10

NOSS SPACECRAFT - DEPLOYED CONFIGURATION - SIDE VIEW - SHADY SIDE



NOSS SPACECRAFT

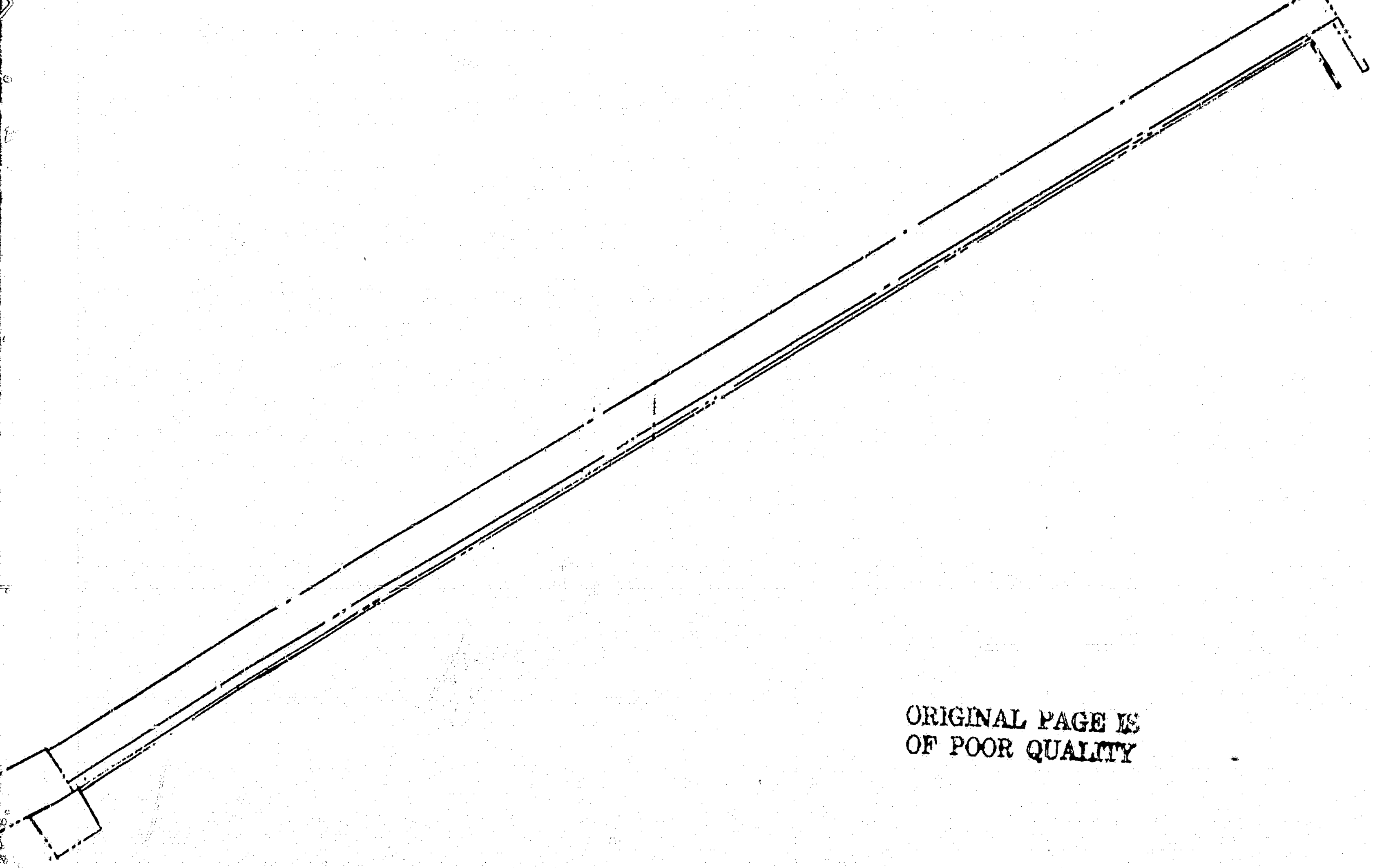


FIGURE 7-11
NOSS SPACECRAFT - DEPLOYED CONFIGURATION - END VIEW

FOLDOUT FRAME 2

C-3